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ADVANCED PLANETARY STUDIES
FOURTH ANNUAL REPORT

by

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for

Lunar and Planetary Programs Division
Office of Space Science
NASA Headquarters
Washington, D.C. 20546

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FOREWORD

This report summarizes the results of advanced studies and planning support performed by Science Applications, Inc. (SAI) under Contract No. NASW-2893 for the Lunar and Planetary Programs Division, Code SL, of NASA Headquarters during the twelve month period 1 February 1976 through 31 January 1977. A total effort of 9233 man-hours (57 man-months) was expended on five specific study tasks and one general support task. The total contract value was \$257,249, with 87% of the work performed by the staff of the SAI Chicago Office. Inquiries regarding further information on the contract results reported herein should be directed to the study leader, Mr. John Niehoff, at 312/885-6800.

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1. INTRODUCTION

Science Applications, Inc. (SAI) participates in a program of advanced concepts studies and planning analysis for the Lunar and Planetary Programs Division, Code SL, of NASA Headquarters. SAI's charter is to perform preliminary analyses and assessments for Code SL planning activities. Specifically, the objective of this support is to ensure NASA with an adequate range of viable future planetary mission options such that its objective of solar system exploration can be pursued in an effective manner within the changing constraints of our Space Program. The nature of the work involved is quite varied, ranging from fast response items to pre-Phase A level mission studies. During the past contract year, a total of twelve SAI staff members contributed to this effort.

The purpose of this Annual Report is to summarize the significant results generated under this advanced studies contract during the twelve month period, 1 February 1976 through 31 January 1977. Progress reports on the task efforts are given at scheduled quarterly reviews. Task reports are prepared at the completion of each task and presentations of significant study results are given to a wide audience at NASA Headquarters, NASA Centers, and at technical meetings. This report, therefore, is necessarily brief. The intention is to direct previously uninformed, but interested, readers to detailed documentation and to serve as a future reference to completed advanced studies.

Each of the six contract tasks are presented in the next section. A brief description is given of the analyses performed along with key results and conclusions. The final section of the report contains a bibliography of the reports and publications that have resulted from these task analyses. SAI is presently beginning a new 24-month period of advanced studies with Lunar and Planetary Programs Division. A schedule of eleven tasks is planned for this period, several of which are continuing research on work reported here.

2. TASK SUMMARIES

A schedule of six study tasks was planned for the twelve month contract period, 1 February 1976 through 31 January 1976. The sixth task, a performance analysis of Venus Surface Sample Return Missions, was replaced by a new task aimed at defining planetary mission discriminators on the choice of solar electric or solar sailing for interplanetary low-thrust propulsion. This adjustment was made in support of a rapidly evolving NASA requirement to develop low thrust propulsion for early 1980 mission opportunities. The Venus surface sample return problem was rescheduled for consideration in the next contract period. The six tasks, then, which were studied are:

- 1) Advanced Planning Activities
- 2) Cost Estimation Research
- 3) Planetary Missions Performance Handbooks
- 4) Penetrator Advanced Studies
- 5) Mercury Mission Transport Study
- 6) SSEP/SAIL Discriminators Definition

Task 1, Advanced Planning Activities, is a general support task designed to provide a budgeted level of effort for technical assistance on short-term planning problems which occur daily within the Lunar and Planetary Programs Division. The remaining five tasks are planned efforts with specific objectives of analysis.

A total of 9233 man-hours (57 man-months) was expended in completing the task schedule. A summary description and discussion of key results for each task are presented in the subsections which follow. The level of effort devoted to each task is given with the task title at the beginning of each subsection. Specific reports generated for each task as part of the contract are noted in the list of publications to be found in Section 3 of this report.

2.1 Advanced Planning Activity (3134 man-hours)

The purpose of this task is to provide technical assistance to the Lunar and Planetary Program Division on unscheduled planning activities which arise during the contract period. This type of advanced planning support is a traditional segment of the broader studies work the staff at SAI have performed for Code SL during all past contract periods. Subtasks within this activity range from straightforward exchanges of technical data by phone, through multi-page responses by mail or telecopier, to more extensive memoranda and presentations, and occasionally to complete status reports on subjects of particular interest. The level of effort per subtask can vary from as little as one man-hour to as much as three man-months. A total of 26 of the more significant advanced planning subtasks, performed during the recently completed contract period, are summarized here. Each of these was the subject of a written submission at the time of its completion. Descriptive titles of these subtasks are tabulated in chronological order in Table 1. A brief summary of each of these subtasks is presented in the subsections which follow.

2.1.1 Execliptic Mission Planning

The purpose of this subtask was to update execliptic mission options data sent to Dr. Simpson of the University of Chicago last year for the purpose of a review paper on execliptic mission planning. Characteristics of the current baseline dual-launch Jupiter swingby execliptic mission profile and two single-launch back-ups were collected, compiled, and forwarded to Dr. Simpson with a memo of explanation. Included in the package was an explanation of the Δ VEGA* technique of energy magnification for interplanetary transfers. Time and reliability penalties required to achieve the Δ VEGA energy gain were also discussed.

* Δ VEGA: Δ V Earth Gravity Assist

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TABLE 1
SUMMARY OF 1976-77 ADVANCED PLANNING ACTIVITY

Subtask	Dates	Subject Title	Submitted To
1	Feb. 1976	Execiptic Mission Planning	Simpson/U of C
2	Feb. 1976	Shuttle Launch Capabilities Guideline Statement	COMPLEX/SSB
3	Feb. 1976	Summary of Special Solar System Mission Opportunities	Code SL/NASA
4	Feb. 1976	Ballistic Planetary Program Models for 1980's	Code SL/NASA
5	Feb. 1976	Comparison of Titan/IUS and Titan/Centaur Launch Vehicle Performance	Code SL/NASA
6	Mar. 1976	Missions to Asteroid 1976AA	Shoemaker/CALTECH
7	Apr. 1976	Advocacy Statement Review	Code SL/NASA
8	Apr. 1976	Planetary Mission Opportunities Summary	Herman/Code SL
9	May 1976	Coordinated JOp/Jex Jupiter Encounters	Van Allen/U of I
10	May 1976	NASA Planetary Advocacy Statement	Code SL/NASA
11	June 1976	Missions to Asteroid 1943	Shoemaker/CALTECH
12	June 1976	Low-energy Shuttle Transition Period Mission Opportunities	Herman/Code SL
13	June-July 1976	NASA 5-Year Planning Support	Code SL/NASA
14	July 1976	Shuttle Payloads Economics Analysis Support	Code SL/NASA
15	Aug. 1976	Cost Estimation Support of Mars Strategy Planning	Code SL/NASA
16	Sept. 1976	Launch Vehicle Performance Requirements for the Planetary 5-Year Plan	Code SL/NASA
17	Sept. 1976	Presentation of Penetrator Applications and Feasibility	COMPLEX/SSB
18	Oct. 1976	Penetrator RTG Specifications	Code SL/NASA

TABLE 1 (Continued)

SUMMARY OF 1976-77 ADVANCED PLANNING ACTIVITY

Subtask	Dates	Subject Title	Submitted To
19	Oct. 1976	Presentation of Mars Penetrators and Hard Landers	TBSWG
20	Nov. 1976	Preliminary Summary of Mars Follow-On Options	COMPLEX/SSB
21	Nov. 1976	Planetary Launch Cost Support of Shuttle LCC Analysis	Code SL/NASA
22	Dec. 1976	Viking Follow-On Mars Mission Options Presentation	Herman/Code SL
23	Dec. 1976	Reestimation of Planetary 5-Year Plan Mission Costs	Code SL/NASA
24	Jan. 1977	ARC Penetrator Cost Estimate Appraisal	Code SL/NASA
25	Jan. 1977	Mars Mission Options Presentation	MSWG
26	Sept.-Jan 1977	Planetary Opportunities Calendar	Code SL/NASA

2.1.2 Shuttle Launch Capabilities Guideline Statement

A reference launch vehicle capability graph was prepared at the request of the COMPLEX/SSB for planetary missions during the period 1981 to 1985. On a plot of escape payload versus escape energy, low, medium and high energy mission points were set as prescribed by COMPLEX. These were 7000 kg @ $C3 = 0 \text{ km}^2/\text{sec}^2$, 2000 kg @ $C3 = 90$ and 500 kg @ $C3 = 150$, respectively. Also shown on the graph for comparison was the launch performance of the present Titan IIIE/Centaur D1-T/TE 364-4 vehicle. Its capability is below that of all three design points identified above. A finished copy of the graph was forwarded to the COMPLEX for inclusion in their report to the SSB.

2.1.3 Summary of Special Solar System Mission Opportunities

A summary of unique mission opportunities during the period 1973-85 was requested by Code SL for the purpose of reviewing how such opportunities have recently been missed and for making a case for better program planning in the future. A total of 14 such opportunities were identified and briefly described with regards to uniqueness, relevance to planetary exploration planning, and comparison of characteristics with generally available opportunities (if any) to the same targets. Targets covered by this survey include comets, asteroids, and the outer planets. It can safely be said at this point, that nine of the 14 opportunities identified already have been or will be missed.

2.1.4 Ballistic Planetary Program Models for the 1980's

A set of 7 inner planet missions, 5 outer planet missions, and 3 small body missions was specified for program modelling for the period 1981-1994. All missions required ballistic interplanetary transfers. Three program scenarios with launch frequencies of >1 , ~ 1 , and <1 launch per year were also given. The purpose of this exercise was to investigate program funding requirements for the 1980's at three levels of activity, assuming only existing propulsion technology.

Cost estimates were collected and/or generated for each of the 15 missions. Adding the costs of current program run-outs, waterfall charts were prepared for three program scenarios and submitted to Code SL/NASA HQ for study along with the individual mission costs. Peak funding for the scenarios reached approximately \$450M, \$340M, and \$230M, respectively, in real year dollars.

2.1.5 Comparison of Titan/IUS and Titan/Centaur Launch Vehicle Capabilities

The purpose of this task was to prepare a graphical performance comparison of various Titan IIIE/IUS vehicle configurations against reference Titan IIIE/Centaur D1-T/TE 364-4 capability. The intent was to be able to consider the credibility of using the Titan IIIE in combination with developed IUS propulsion to meet transition period mission launch requirements in the event of a slip in the Shuttle test program. Performance curves were prepared using data generated by Battelle Columbus Laboratories for NASA Headquarters for the following options:

- 1) Titan IIIE/IUS (II)
- 2) Titan III/IUS (III)
- 3) Titan IIIE/IUS (IV)
- 4) Titan IIIE(7)/IUS (IV)

None of the options, including the fourth case which uses a non-existing 7-segment Titan III configuration with a four-stage IUS was able to equal the injected mass performance of the Titan IIIE/Centaur D1-T/TE 364-4 reference vehicle. Hence, mating the IUS with the Titan IIIE would not meet all planetary transition mission launch requirements in the event of a slippage in the Shuttle IOC date.

2.1.6 Missions to Asteroid 1976AA

Performance analyses of round-trip missions to the newly discovered earth-crossing asteroid 1976AA were conducted. Both manned

and unmanned missions were considered over an opportunity span equivalent to the object's synodic period with the earth, i.e., about 19 years. The purpose of this short study was to determine if reasonably low round-trip energy requirements might exist for this near-earth object, thus enabling a possible early mission. Optimum one-way data, already generated by Bender of JPL, were used to help locate best round-trip opportunities in the synodic cycle. The performance results showed that both unmanned and manned round-trip energy requirements were very high, even in the best years, due to 1976AA's high inclination of 19° to the ecliptic plane. For example, it would require at least 28 Shuttle launches to assemble the hardware components in low-earth orbit for a 365-day manned round-trip mission launched in 1993.

2.1.7 Advocacy Statement Review

Early in the contract period, NASA Headquarters undertook an activity to strengthen its advocacy of space science and exploration. This was the first of several subtasks (e.g., see 2.1.10 below) supporting that activity. Its purpose was to review and critique an initial general advocacy package outline, generated by the Associate Administrator for Space Science. The package included the following elements:

- 1) Problem, Approach and Supporting Material
- 2) Need for Basic Science
- 3) NASA Role in Science
- 4) Exploration Themes
- 5) Exploration Elements
- 6) Implementation

Each of these was reviewed for content, and comments and questions were returned to Code SL both regarding stated rationales and guidelines for subsequent work on the package.

2.1.8 Planetary Mission Opportunities Summary

The purpose of this subtask was to prepare a set of viewgraphs summarizing planetary mission opportunities during the 10-year period of

1975-85 for a talk by the Manager for Advanced Programs and Technology (Code SL) to be given at a meeting of the Solar System Science Working Group (Code ST). A total of 16 figures and tables were prepared divided into two groups: 1) inner solar system, and 2) outer solar system. For the inner solar system group, orbiter performance at Mercury, Venus and Mars was presented along with Mars sample return mission capabilities. Several comets and multi-asteroid flyby missions were also presented. For the outer solar system, orbiter mission performance was summarized at Jupiter and Saturn, and payload/flight characteristics of Uranus, Neptune and Pluto flyby mission using Jupiter and/or Saturn swingbys was generated. Both ballistic and low-thrust flight modes were considered.

2.1.9 Coordinated J_{O_p}/J_{ex} Jupiter Encounter

The purpose of this subtask was to investigate the planning requirements of the Jupiter Orbiter/Probe (J_{O_p}) and Jupiter Execliptic (J_{ex}) missions such that a simultaneous 3-spacecraft encounter of Jupiter might be possible leading to enhanced magnetospheric mapping of the giant planet. One J_{O_p} spacecraft and two Jupiter-assisted polar execliptic spacecraft are expected to be launched in the early 1980's. The problem is one of coordinating the two projects such that all three spacecraft can be favorably positioned in Jupiter's magnetosphere at the same time without unduly constraining their mission-specific objectives. Assuming a J_{O_p} launch in 1981/2, acceptable strategies were found for two cases: 1) night-side entry of the J_{O_p} probe, and 2) day-side entry. When the J_{O_p} probe uses a night-side entry and the J_{ex} launches are in 1983, all three spacecraft are at Jupiter in May 1984. For the day-side entry case, the J_{ex} launch must be delayed one year to 1984, with all three spacecraft then being at Jupiter in June 1985. Results of these strategies were mailed to Dr. James Van Allen at the University of Iowa, and several members of the J_{O_p} Science Working Group.

2.1.10 NASA Planetary Advocacy Statement

A group of scientists and engineers was assembled by the Code SL Manager for Advanced Programs and Technology for the purpose of generating an advocacy statement on solar system exploration as a contribution to the Spring 1976 NASA Advocacy Activity. This subtask involved SAI's participation in that group effort. The group consisted of NASA Headquarters, NASA Center, JPL and SAI personnel. Several meetings and numerous teleconference calls and telecopied data exchanges were required to complete a draft package on exploration of the solar system. A format of illustrations with facing page descriptors was adopted for the package. The resulting product began with the concerns of mankind, working through subjects of goals, missions and benefits to the concept of an exploration program and finally the program itself. Included in the package were issues of historical perspective, implications of choice, refined goals of exploration, additional program alternatives, and pictorial summaries of significant future exploration endeavors. The draft package, submitted to NASA Management, was eventually published as a brochure entitled "Exploration of the Solar System".

2.1.11 Missions to Asteroid 1943

Following the performance results of round-trip missions to the Apollo asteroid 1976AA (see 2.1.6 above), it was requested that a similar analysis be performed for the Amor object 1943 (originally identified as 1973EC) which had a much lower inclination, less than 9° , and would hopefully have lower mission energy requirements. Again, round-trip trajectory data were generated for all the opportunities (five) within one syzygetic period (12 years) to find the optimum round-trip requirements. Constrained (one-year) and unconstrained trip times were considered. The performance results revealed an exceptionally low-energy, unconstrained round-trip mission with a launch opportunity occurring in 1992. A round-trip time of just under 3 years, however, probably restricts this mission to an unmanned configuration. Yet the

entire mission could be flown ballistically with one Shuttle launch returning a 1.0 kg sample of the asteroid to the earth. Constraining the total trip time to one-year and adding life support hardware for a manned mission raised the Shuttle requirement to 23 launches with the stay time severely restricted to 10 days. It was concluded that 1943 was not a good target for an early manned asteroid mission.

2.1.12 Low-Energy Shuttle Transition Period Mission Opportunities

This subtask involved the preparation of several viewgraphs to be added to a summary presentation by the Code SL Manager for Advanced Programs and Technology to the COMPLEX on the subject of Shuttle launch capabilities for planetary missions in the 1980's. The prepared material dealt with low-energy mission opportunities, requiring no more than a Thor/Delta or Atlas/Centaur launch vehicle, which might be flown in the event of a Shuttle IOC Date slippage. Launch capabilities and mission payload requirements were matched for seven low-energy missions. The list included one Venus mission, three Mars missions, one comet mission, and two asteroid mission. Comments on mission constraints specific to limited launch capability were also provided. A mission capture graph for the Shuttle/IUS was also provided to indicate where degradation in forecasted launch performance would have its first effects on planned planetary missions.

2.1.13 NASA 5-Year Planning Support

This subtask was a two-month support activity related to Code SL's first annual 5-year planning exercise. The purpose of the exercise was to synthesize the many planning activities continually in progress at NASA into a realistic near-term plan which is consistent with anticipated funding and serves as a guide for future planning activities. Support analyses on this subtask included project manpower and cost estimates, estimate revisions to accommodate both inheritance and mission scope factors, and mission integration into a 5-year plan. Cost estimates

were worked in both fixed and real year dollars. Programmatic results were presented in waterfall chart formats to be compared against anticipated funding guidelines. Numerous iterations on project start dates required repeated recomputations of project cost spreads for resource planning. It is planned to repeat the activity annually, each time adding a new year to the plan and dropping the just completed one.

2.1.14 Shuttle Payloads Economics Analysis Support

The purpose of the subtask was to provide Code SL with estimated project resources requirements for planetary missions planned through 1991. The results were needed by the the Shuttle Payloads Office which was involved in a economic analysis of Shuttle payload loading. A total of 13 missions were included in a typical program plan which was built up from the 5-Year Plan results (see 2.1.13) and cost data. Several new missions including an Encke Rendezvous, a Mars Surface Sample Return, and Jupiter-Swingby/Pluto Flyby mission were added to complete the plan. Cost spreads in real year dollars were then computed and integrated to give an annual cost profile of the plan. Peak annual funding of \$738M occurred in 1982, due largely to MSSR costs. A detailed breakdown of costs by mission and fiscal year was forwarded to Code SL for subsequent inclusion in the Shuttle payload planning exercise.

2.1.15 Cost Estimation Support of Mars Strategy Planning

Mars mission cost estimates were developed for several exploration strategies classified as strong, nominal and weak responses to the Viking mission success. These estimates were made as part of a larger Code SL exercise to assess its position on an early new initiative to Mars after Viking. Cost estimates were generated to Penetrator, Orbiter/Penetrator and Sample Return missions. Dual launch sample return missions were considered with and without rovers. The sample return mission with the rover carried along, returned a small 1.0 kg sample. Without

the rover, it was assumed that previous rovers had collected a larger sample of 20 kg which was to be retrieved and returned to the earth. Project element costs as well as totals were prepared for consideration both by NASA Headquarters and JPL.

2.1.16 Launch Vehicle Performance Requirements for the Planetary 5-Year Plan

The purpose of this subtask was to redetermine launch vehicle injection points (escape payload, C3) for those missions in the newly developed 5-Year Plan which were changed or were new additions to the existing advanced studies data base. These mission injection points are necessary for analyzing Shuttle/Upper Stage capture capabilities. Two missions in the 5-Year plan required updating. Those missions and their revised injection points were as follows:

Mission	Flight Mode	Escape Payload (kg)	C3 (km ² /sec ²)
Venus Orbital Imaging Radar	Ballistic	3750	14
Saturn Orbiter w/Titan Lander	SEP	4950	16

2.1.17 Presentation of Penetrator Application and Feasibility

A review presentation of penetrator application and feasibility studies was presented to COMPLEX as a status report on this concept for planetary surface exploration. The Mars penetrator design was reviewed followed by a summary presentation of penetrator design requirements and capabilities at ten different solar system targets. It was apparent that although the penetrator design might be applicable to many targets (with some subsystem modifications, particularly

thermal control and power), it was just as apparent that the required delivery systems from spacecraft to surface were not. The Mercury Alternate Lander design was presented and compared to Mercury penetrators. The Alternate Lander has definite advantages in surface (versus subsurface) science capacity and in lifetime over the penetrator. Penetrator development issues related to science, rough landers, instruments, deployment techniques, thermal control, data storage, lifetime, radiation hazard, and earth applications were presented. The question of mission sequencing of penetrator applications was briefly discussed citing the priority of design requirements implied by the order of missions flown. Finally, a synopsis of the penetrator program status was given to the COMPLEX as a point of departure for their deliberations.

2.1.18 Penetrator RTG Specifications

Preliminary specifications for penetrator RTG development was requested by NASA Headquarters for planning purposes. A table of specifications was prepared after consultation with ARC/NASA personnel currently participating in the penetrator advanced technology program. Parameters specified included End-of-Life, electrical and thermal power output, fuel, weight, packaging, shock, and shielding. Comments were added to explain and/or qualify the specifications. A copy of the completed table was sent to ARC/NASA as well as Code SL.

2.1.19 Presentation of Mars Penetrators and Hard Landers

The purpose of this subtask was to prepare and present to the Terrestrial Bodies Science Working Group possible options for deployment of penetrators or hard landers at Mars. The characteristics of lander deployments on a 1981 Mars orbiter mission were reviewed including a baseline profile for the orbiter phase of the mission. Geometrical constraints associated with direct entry, elliptic orbit deployment and circular orbit deployment of landers was discussed. Deployment scenarios for penetrators and hard landers were presented. The mass requirements associated with each of these scenarios was developed and compared with Shuttle/IUS launch capabilities.

All scenarios (up to nine penetrators or four hard landers) required the IUS (III), but were easily performed with this configuration. The presentation was concluded with a proposed development schedule for a 1981 launch assuming an FY 1979 project start.

2.1.20 Preliminary Summary of Mars Follow-On Options

The purpose of this subtask was to investigate and present a preliminary summary of feasible options for a 1981 Mars follow-on mission. All options included the Mars Polar Orbiter as a design base; the additional options considered were penetrators, hard landers, a mobile lander, and a mobile lander plus penetrators. Each of these options was discussed with the COMPLEX touching on such subjects as deployment strategies, impact site accessibility, spatial coverage, resolution, telecommunication capability, launch vehicle reserve, and project cost. Data were presented in a comparative fashion so that assessments could be made of relative capabilities and requirements for each option.

2.1.21 Planetary Launch Cost Support of Shuttle LCC Analysis

This subtask was performed in response to a request from the Office of Space Flight to Shuttle users for relaunch costs associated with either a missed opportunity or a launch failure. A total of eight planetary missions for the period 1981-91 were analyzed for add-on costs due to launch problems. Each mission had to be individually analyzed because project spare hardware philosophy and fall-back launch opportunity characteristics were continuously variable across the mission set. Supporting rationale for the assumed work around plans associated with these costs was provided along with the individual cost data.

2.1.22 Viking Follow-On Mars Mission Options Presentation

Six Mars mission options were analyzed in this subtask in preparation for a presentation by Code SL's Manager for Advanced Programs and Technology to the Physical Sciences Committee. These mission

options were as follows:

- 1) Polar Orbiter
- 2) Polar Orbiter/Penetrators (6-9)
- 3) Polar Orbiter/Rough Landers (4)
- 4) Orbiter/Mobile Lander
- 5) Polar Orbiter/Mobile Lander
- 6) Polar Orbiter/Mobile Lander/Penetrators (3)

Beginning with the data base generated in Subtask 2.1.20 (see above) each option was analyzed for an operations profile, orbit parameters, and propulsion parameters.

A further comparison was prepared within each option by considering both 1981 and 1984 launch opportunities. A summary viewgraph was prepared for each option showing all comparison data. A final viewgraph summarizing the performance conclusions of the comparison was also prepared. Key conclusions were that only Option 1 could be performed by the IUS (II), that Options 5 and 6 required complete IUS (III) capability and a new retro propulsion design (still earth-storable, however), and that 1984 is a more difficult opportunity from the viewpoint of performance requirements than 1981.

2.1.23 Reestimation of Planetary 5-Year Plan Mission Costs

The purpose of this ssubtask was to review and reestimate as necessary project costs of missions included in the earlier 5-Year planning exercise (see subtask 2.1.13 above). Many of the estimates done during the exercise were made either as extrapolations from existing mission data or with very preliminary mission definitions. With the pressure to complete the plan on a short time scale past, Code SL decided it would be wise to reexamine its estimates in a more deliberate and thorough manner. Costs were recomputed for eight of the planetary missions in the plan and were reduced for Code SL into project categories and project cost spreads. The results were forwarded to NASA Headquarters

to form an improved data base for the next 5-Year planning cycle, scheduled for March 1977.

2.1.24 ARC/NASA Penetrator Cost Estimate Appraisal

The purpose of this subtask was to compare a cost estimate of Mars penetrators recently computed by ARC/NASA with earlier pre-Phase A estimates generated by SAI for planning purposes. The ARC estimate totaled \$69.3M for six flight articles and one spare in FY '79 dollars. The breakdown on this total included \$6.7M for operations, \$18.1M for science, and \$44.5M for design/development/manufacture of the engineering subsystems. SAI's estimates had only been made for this last cost element, i.e., engineering subsystems. Rerunning these estimates for the 1981 mission yielded values of \$27.8M for one flight unit plus a spare and \$4.1M for each additional flight article, again in FY '79 dollars. Hence, a subtotal value for six penetrators plus one spare of \$48.2M was computed, which compares favorably with the \$44.5M ARC figure.

2.1.25 Mars Mission Options MSWG Presentation

This subtask was a presentation of the material developed on Viking follow-on mission options at Mars (see subtask 2.1.22 above) to the first meeting of the Mars Science Working Group. Preparation and presentation of the material was done by the SAI Advanced Studies leader. Emphasis in the presentation was on a comparison of the 1984 opportunity options. The data were also compared to new alternatives presented by JPL at the same meeting.

2.1.26 Planetary Opportunities Calendar

This was a major subtask of the advanced planning activity during this contract period taking approximately four months to complete. The purpose of the Calendar is to provide an overview of launch windows to solar system targets through the end of the century for program planning purposes. Opportunity data were prepared for the eight major targets and for selected comet missions in the period 1980-2000. A wide range of flight modes was considered in generating these data. The direct

ballistic option was included for all of the major bodies except Mercury. In other cases for which this option was not a realistic mission alternative, it served as a reference for comparison with a variety of indirect gravity-assisted swingby modes. These include the recently studied VEGA/SEEGA* options which utilize an Earth reencounter to set up the final trajectory leg, effecting a considerable savings in energy requirements over the direct outer planet transfers. Data for these options was also included, to the extent to which it was currently available. Flight modes presented for Mercury included both ballistic Venus swingby and solar electric low-thrust. Type I and II transfers were considered for Venus and Mars, and a dual launch version of the Mars sample return mission was briefly analyzed for the nine opportunities available. The results of this effort were assimilated in a calendar style format with one page of performance summaries and one page of opportunity dates presented for each target. A summary of the opportunities presented in the calendar is given in Table 2 as a matrix of target and flight mode versus launch year. The matrix is nearly complete, the exception being the VEGA flight mode for which opportunity data are not yet available beyond 1991.

*VEGA/SEEGA: Venus Earth Gravity Assist/Solar Electric Earth
Gravity Assist

Table 2

PLANETARY MISSION OPPORTUNITY MATRIX

CALENDAR LAUNCH YEAR	MERCURY		VENUS		MARS		JUPITER		SATURN		URANUS		NEPTUNE		PLUTO	
	BALLISTIC		SEP		DIRECT		VEGA		DIRECT		J/U		S/U		VEGA	
	116	584	780	399	378	2015	370	2006	2026	367	2002	367	2001	2001	2001	2001
Interval (days)	116	584	780	399	378	2015	370	2006	2026	367	2002	367	2001	2001	2001	2001
1980	X	X		X	X		X	X	X	X	X	X	X	X	X	X
81	X	X	X		X		X	X	X	X	X	X	X	X	X	X
82																
83	X	X		X	X		X	X	X	X	X	X	X	X	X	X
84																
85	X	X	X	X	X		X	X	X	X	X	X	X	X	X	X
86	X	X	X	X	X		X	X	X	X	X	X	X	X	X	X
87																
88	X	X	X	X	X		X	X	X	X	X	X	X	X	X	X
89	X	X	X	X	X		X	X	X	X	X	X	X	X	X	X
90																
91	X	X	X	X	X		X	X	X	X	X	X	X	X	X	X
92																
93																
94	X	X	X	X	X		X	X	X	X	X	X	X	X	X	X
95																
96	X	X	X	X	X		X	X	X	X	X	X	X	X	X	X
97																
98																
99	X	X	X	X	X		X	X	X	X	X	X	X	X	X	X
2000	X	X	X	X	X		X	X	X	X	X	X	X	X	X	X
2001																

Next opportunities begin in:

2.2 Cost Estimation Research (1716 man-hours)

Cost estimation analysis has been an on-going Advanced Studies support task for four years. Its objective is to develop and implement a methodology for estimating costs of future lunar and planetary flight projects. Its purpose is to provide reasonably accurate cost estimates, based on pre-Phase A study definitions, to key advanced planning activities within the Lunar and Planetary Programs Division. A flight project cost estimation model has been in existence at SAI for the past three years as a result of this task effort, and has been regularly improved and expanded in scope of application as a result of this on-going research. The nature of the work falls into one of three general subtasks:

- 1) Flight Project Data Collection,
- 2) Modelling Analysis,
- 3) Cost Estimation.

Work is done in all three subtask areas each year. The level of effort expended on data collection has stabilized during the past several years with three to four flight projects being tracked at any given time. There has been a shift in emphasis, however, within the other tasks with increasingly more effort expended now on applications and less on modelling. This may occasionally change as new features are added to the cost model, but generally emphasis should continue on applications. Each of the subtasks is briefly summarized in the following subsections.

2.2.1 Flight Project Data Collection

Historically, estimates of future flight project costs have frequently been underestimated by substantial amounts. One of several reasons for this situation has been the lack of an adequate data base from which to judge new endeavors. A second cause has been failure to take into consideration capabilities and requirements fostered by new technologies. These problems emphasize the importance of two attributes of an acceptable data base, i.e., breadth and currency. Neither of the attributes can be achieved and maintained without continuous diligence and care.

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Such is the case with the SAI data base. Since data collection began more than four years ago, every effort has been made to incorporate all relevant lunar and planetary flight project data into it. This means collection of Level Three or better Work Breakdown Structure data on anywhere from quarterly to annual periods depending on the project maturity. Direct Labor, burden, materials and miscellaneous costs must be tracked on every element of the projects. These data are then reduced into new categories consistent with modelling algorithms used in the cost model.

During the 1976-7 contract period, new data was collected and reduced on three flight projects: Viking Orbiter, Viking Lander, and the Mariner Jupiter/Saturn (MJS) missions. As a result of this effort, the Viking project costs are virtually complete, whereas the MJS data is now about 60% complete with the remaining expenditures in this project being estimates-to-complete. The complete SAI cost data base currently consists of ten lunar and planetary flight projects undertaken during the period 1962 to present. Data collection/reduction activities in the coming contract period will focus on continued collection of MJS data, and initial acquisition of Pioneer Venus costs, which will include the first flight project data for atmospheric probes. The Jupiter Orbiter flight project, presuming new start approval for FY '78 is also a near-future addition which will reflect the first use of NASA standardized components in flight project hardware.

2.2.2 Cost Modelling

The cost modelling subtask's initial objective was the development of a flight project cost estimation analog whose input requirements could be restricted to pre-Phase A level mission definitions. Such a cost model, using direct labor hours as the working cost parameter, has been developed at SAI and is actively in use. The on-going purpose of this subtask is to refine and expand the model's scope of application as permitted by the expanding base of flight project data resulting from the effort expended in the previous subtask.

Development of the cost model was initiated with the re-distribution of flight project cost data into a minimum set of categories,

each of which was to be modelled as a function of some pre-Phase A mission parameter(s). The categories found to be most acceptable for this purpose fell naturally into two classes: 1) subsystem hardware costs which have both non-recurring and recurring elements and 2) project support costs which are recurring elements scaled (in part) to the magnitude of total hardware costs. The specific categories used are as follows:

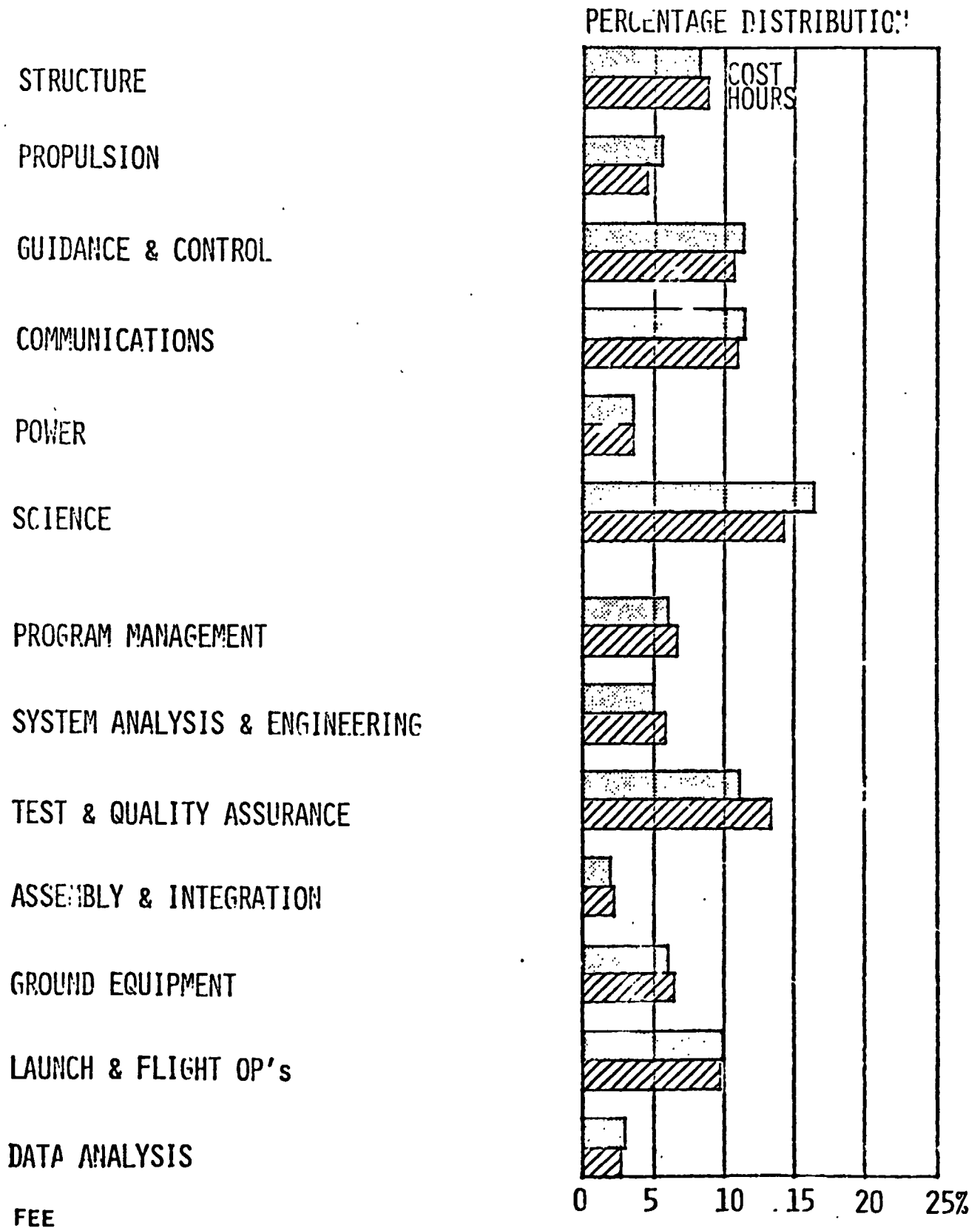
1) Hardware Categories
Structure
Propulsion
Guidance & Control
Communications
Power
Science

2) Support Categories
Program Management
Systems Analysis & Engr.
Test & Quality Assurance
Assembly & Integration
Ground Equipment
Launch & Flight Ops.
Data Analysis

An obvious dependent parameter choice for modelling the costs of these categories is dollars. However, the use of dollars often obscures the real cost because of wage inflation factors, overhead rates, fees, etc. Planetary missions are typically characterized by very low production volume and high development costs, i.e., they are labor-intensive endeavors. Hence, the use of direct labor hours was considered as a possible alternative to dollars. Productivity rather than wage rate (and hence inflation factors) becomes a key measure of cost when using direct labor hours. Also, direct labor is a common denominator of NASA cost reporting requirements from which overhead, G&A and fee are computed. Of concern in the use of direct labor hours was the omission of project materials costs. To examine how well direct labor alone could track total project cost, comparisons are continually made between cost per category and direct labor per category. For both parameters percentage comparisons, averaged over the entire ten project data base, are shown in Figure 1 for each category defined above. The comparison validates the credibility of direct labor hours to adequately track total project cost. Further analysis of the data base also revealed that direct labor hours represent 30% of total flight project

FIGURE 1

PERCENT COMPARISON OF DOLLARS* AND LABOR HOURS
ALL MAJOR PROJECTS (AVG)



cost with only a few percent variation over the entire data base. It was concluded the labor hours are indeed a very good parameter of cost, and further that modelling project direct labor is essentially equivalent to modelling total planetary flight project costs.

The choice of direct labor hours to model cost opened the way for the actual modelling analysis. Labor estimating relationships (LER's) were developed for each cost category. The non-recurring direct labor hours (NRDLH) of the hardware categories were modeled first since they were most readily associated with pre-Phase A mission parameters, particularly weight. Recurring direct labor hours (RDLH) were modeled next as a function of the NRDLH and number of flight articles. Pre-launch support category direct labor hours were modeled as a function of the accumulated total hardware direct labor hours. Launch and post-launch functions were modeled from pre-Phase A mission parameters, particularly event times, as well as accumulated direct labor hours.

A flow chart depicting the total estimation procedure is presented in Figure 2. The heavy arrows indicate the primary flow of the estimation process using the various LER's outlined above. Both hardware and support category direct labor hours (DLH) are converted to dollars using modeled category wage rates and inflation factors consistent with the anticipated flight project period. These costs are accumulated to a total direct labor (DL) project cost which is then ratioed up (\div by 30%) to finally determine total project cost. Note that inheritance (cost saving) factors can be added to the input stream at the hardware cost level to reduce required NRDLH levels for subsystem development. Inheritance is considered as a percentage of each category which qualifies for cost savings with actual savings accrued as many as three levels of inheritance. Reductions in hardware NRDLH are allowed to ripple through the estimation procedure so that additional savings are also realized in associated support categories. The inheritance method is sufficiently general to permit eventual inclusion of standardized hardware cost benefits when such data become available from flight project experience.

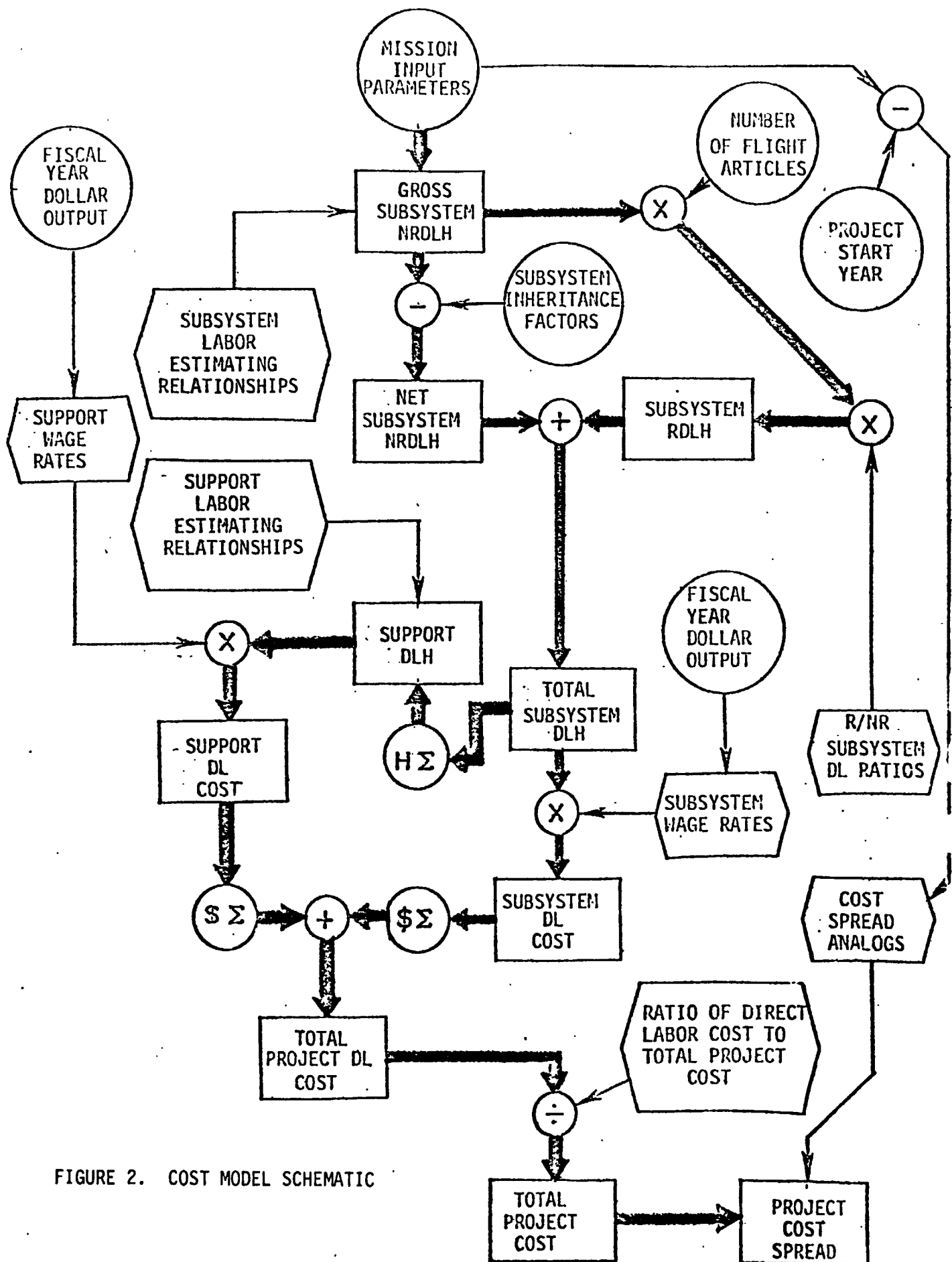


FIGURE 2. COST MODEL SCHEMATIC

Both the LER's and their synthesis into an estimation procedure are the subjects of the continued analysis of this subtask. As a result of this on-going effort the cost model is now applicable to a wide scope of mission concepts including flybys, orbiters, entry probes, landers, and sample returns. Subtask analysis is currently focused on improving entry probe cost estimates with results not yet complete as Pioneer Venus flight project data are still being collected. As the model has been expanded and improved so also have the input requirements increased. The current list of possible input parameters is presented in Table 3. This list will undoubtedly continue to grow with further model improvements, but will be diligently constrained to a pre-Phase A study information level.

Cost model accuracy objectives are twofold: 1) Estimates of total costs for projects included in the data base should not differ from actual by more than 10%; 2) New project estimates should not be in error by more than 20% with mission scope held constant. Error analysis of the model against the data base presently shows a mean error of -6.4% in cost (i.e. underestimating) with a mean absolute error of 12.9%. Applications to date against existing programs not in the data base indicate that errors for new flight projects are probably not greater than 25%.

2.2.3 Applications

Applications of the cost model have continued to increase with its refinement and expanding scope. During the past contract period, the model was used extensively in support of advanced planning activities by the Lunar and Planetary Programs Division. Seven of the 26 Advanced Planning Subtasks reported above in Section 2.1 involved cost estimation analyses. These subtasks were as follows:

- 2.1.4) Ballistic Planetary Program Models for 1980's
- 2.1.13) NASA 5-Year Planning Support
- 2.1.14) Shuttle Payloads Economics Analysis Support
- 2.1.15) Cost Estimation Support of Mars Strategy Planning
- 2.1.21) Planetary Launch Cost Support of Shuttle LCC Analysis
- 2.1.23) Reestimation of Planetary 5-Year Plan Mission Costs
- 2.1.24) ARC Penetrator Cost Estimate Appraisal

TABLE 3

COST MODEL INPUT PARAMETERS

- Mission Factors
 - Fiscal Wage Date
 - Date of First Launch
 - Number of Flight Articles
 - Mission Duration
 - Encounter Time
 - Launch Windows
- Structure
 - Total Weight of Structure Subsystem
 - Weight of Mechanisms & Landing Gear
 - Weight of Thermal Control, Pyro & Cabling
- Propulsion
 - Dry Weight of Propulsion System
 - Liquid Vernier Dry Weight
 - Aerodeceleration Subsystem Weight
- Guidance and Control
 - Total Weight of Guidance & Control Subsystem
 - Weight of Radar in G&C Subsystem
- Communications
 - Weight of Radio Frequency Subsystem
 - Weight of Data Handling Subsystem
 - Diameter of Antennas
- Power
 - Weight of Power Subsystem Excluding RTG's
 - Number of RTG Units Per Spacecraft.
 - RTG Fuel Loading (Thermal Watts)
- Science
 - Total Weight of Science Experiments
 - Weight of Lander Surface Experiments
 - Pixels per Line of TV

As an example of the types of data prepared for these activities, a summary of mission costs by project element are presented in Table 4 from the reestimation analysis performed for Subtask 2.1.23. This was the first combined estimation effort of these missions performed under a consistent set of ground rules. Cost spreads on each of these estimates were also generated and are presented in Table 5.

Application of the cost model has also been extended to other contract tasks. It is now used routinely as an estimation tool to add cost data to all advanced mission and concept studies. These added results provide an additional dimension to the evaluation of studies of potential future missions.

TABLE 4

PROJECT COST BREAKDOWNS FOR "5-YEAR PLAN" MISSIONS

Project Categories	Missions							
	VOIR (85)	SU _p (85)	MeO (87)	SO-T _L (88)	AST SURV (88)	JOST-G _L (89)	VO _L (91)	JP (90)
Program Management & Engineering	11.9	7.1	12.2	22.0	5.9	8.8	16.2	3.0
Science								
Instruments	11.7	27.1	42.5	99.3	23.6	29.7	66.4	6.8
Data Analysis	8.1	14.4	10.4	14.5	5.7	10.1	9.1	7.8
Spacecraft								
Bus	114.8	52.8	87.8	58.2	50.4	58.2	66.2	39.5
Probe	-	24.9	-	80.3	-	46.3	71.2	-
Launch & Flight Ops	25.8	33.2	31.0	50.1	17.3	29.0	30.8	19.4
MC ³	5.2	8.1	6.7	10.2	3.7	6.6	5.9	5.0
APA (10%)	17.8	16.8	19.0	33.5	10.7	18.9	26.6	8.1
Sub Total	195.3	184.4	209.6	368.1	117.3	207.6	292.4	89.6
SEP	-	-	20.0	20.0	20.0	20.0	-	-
TOTAL	195.3	184.4	229.6	388.1	137.3	227.6	292.4	89.6

Costs Given in FY 78 Dollars

Amounts in Millions of Dollars

TABLE 5

PROJECT COST SPREADS FOR "5-YEAR PLAN" MISSIONS

Missions	Fiscal Launch Year	Years From Launch										
		-3	-2	-1	0	+1	+2	+3	+4	+5	+6	+7
VOIR	1983	6	52	52	47	38						195
SU _p	1984	21	48	43	27	23	9	1	2	2	4	184
MeO	1987	9	64	64	44	28	21					230
SO-T _L	1988	31	110	100	67	44	14	6	5	6	5	383
AST SURV	1988	3	38	40	29	16	5	4	2			137
JOST-G _L	1989		52	68	47	27	18	11	5			228
VO _L	1991	9	89	85	66	43						292
JP	1990	2	20	20	20	16	3	3	2	2	2	90

Costs Given in FY 78 Dollars

Amounts in Millions of Dollars

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2.3 Planetary Missions Performance Handbook: Vol II-MSSR
Revisions (1196 man-hours)

The purpose of the Planetary Missions Performance (PMP) Handbooks is to provide program planners with the basic performance data essential in the preliminary stages of mission selection and design. In the past, two types of NASA handbooks have been prepared for mission analysis work: 1) raw trajectory data handbooks such as the NASA SP-35 series, and; 2) propulsion system performance handbooks such as the NASA Launch Vehicle Estimating Factors Document. The PMP Handbook series carries performance analysis one step further by combining these two basic groups of data in a form which is directly applicable to mission planning. Typical results show payload mass as a function of flight time, or launch window, as appropriate to the specific mission.

Volume II of the PMP Handbook Series contains missions to the inner planets. The Mars Surface Sample Return mission is treated as a special case, and is allotted a full section of its own. This year's PMP task revised the MSSR section to include dual launch missions based upon currently planned Shuttle/IUS capability. Mass performance summaries are presented for the nine Mars Sample Return opportunities in the period 1980-2000, thus spanning more than a full cycle of Mars launch opportunities (seven in fifteen years).

For each launch opportunity, two single-launch and two dual-launch options are examined. These are: 1) direct atmospheric entry at Mars, or; 2) entry from a specialized landing orbit. Thus, there are four basic mission designs considered, all of which assume rendezvous of a planetary excursion module with an orbiting bus placed in rendezvous orbit. Two other single-launch options which were presented in the previous release of this section are dropped from consideration here. Both of these (direct and orbit entry at Mars) utilize direct return from Mars, via parking orbit. This choice necessitates landing all earth return systems on the planet and, of course, lifting them off again into the parking orbit. With increased dry mass requirements for the Earth Return Vehicle (ERV) - 137 kg versus

the previously assumed 87 kg - these options usually exceed the capability of the baseline three-stage IUS. For example, given other current mass assumptions and propulsion sizing, the larger (137 kg) ERV does not allow useful missions in any of the nine opportunities examined here. Therefore, to accomplish a direct return from Mars with the needed increase in landed mass will almost certainly require the Space Tug. However, such Tug missions would require the entry and landing on Mars of better than twice the mass of similar systems in current design points. Hence, these Tug missions are sufficiently outside the domain of present scaling laws to preclude their consideration here. On the other hand, should increased landed payload be required (e.g., for rovers, or to prepare the way for return of large samples), it can be achieved with the dual launch options, which need not land the earth return systems and which are well within IUS capabilities. Table 6 summarizes the scope of the general mission options and launch selections.

In light of current planning emphasis and sample return science requirements, a baseline mission concept has been chosen for performance cross-comparisons between launch opportunities. Briefly, it calls for two Shuttle/IUS (III) launches on conjunction-class transfers with orbit capture both at Mars and at earth return. A one kilogram sample is to be collected, which serves to size the earth entry capsule and, to a lesser extent, the ERV. Performance summaries of the baseline and several variations upon it are developed for each launch opportunity.

Mars launch opportunities are cyclic in nature - seven opportunities in fifteen years. Thus, the nine opportunities shown are sufficient to investigate the full range of performance. Examination of the hyperbolic velocities at launch and target for both legs (earth-Mars, and Mars-earth) reveals four cyclic patterns - one for each choice of earth and Mars entry modes. These patterns are distinctly out of phase with one another. Consequently, different flight modes for the Mars Sample Return may have different "best" opportunities. Indeed, this turns out to be the case, as will be seen below. For each leg of the journey, the entry mode at the target planet (direct vs. orbit) selects the trajectory to be used, and thus, the impulses

TABLE 6

SCOPE OF MARS SAMPLE RETURN PRESENTATION

• LAUNCH OPPORTUNITIES

1981-----	Nov 81 - Jan 82
1984-----	Dec 83 - Feb 84
1986-----	Mar 86 - May 86
1988-----	May 88 - Jun 88
1990-----	Jul 90 - Aug 90
1992-----	Sep 92 - Oct 92
1994-----	Oct 94 - Nov 94
1996-----	Nov 96 - Jan 97
1999-----	Dec 98 - Feb 99

• MISSION OPTIONS*

o Dual Launch	{ Mars Orbit Capture Mars Direct Entry
o Single Launch	{ Mars Orbit Capture Mars Direct Entry

* All assume Mars Orbital Rendezvous

• LAUNCH VEHICLES

- o Shuttle/IUS (II)
- o Shuttle/IUS (III)
- o Shuttle/Tug (R)/Earth Escape Kick

required. Moreover, the propulsion systems needed to accomplish the return leg must first be carried to Mars on the initial leg. Clearly, the specific mission options selected play as large a role in overall mission performance ability as does variation from one launch opportunity to the next. This is particularly true for single launch cases: slight penalties on the return leg can loom large at earth launch.

Performance of the Mars Surface Sample Return mission can be characterized in many ways. It is useful to know what size sample can be returned, how much mass can be landed upon Mars, and what total launch mass is required. If we assume that these are the three most significant mass figures for a sample return, then it can be said, in general, that the 1986 opportunity is quite poor, and that the early 90's opportunities offer the best possibilities, with 1994 being a banner year. Such a characterization can be misleading: in this case, the 1986 Mars direct entry variation of the baseline mission produces a higher landed mass margin than that available in any other opportunity. This apparent discrepancy is resolved by observing that it is the earth return leg which hurts the 1986 opportunity. (Direct entry at Mars removes the effect of a high orbit capture impulse.) Thus, although it seems unlikely that the baseline mission in 1986 can tolerate much of an increase in returned sample size, the direct entry option in that year can produce a substantial landed mass margin which can be used to accomodate additional surface science or rover systems. Comparative examination of several launch opportunities will expose other tradeoffs in mission performance.

A sample return has so many basic mission parameters which may be subjected to variation in planning exercises that even a handbook must present only a selection of the most interesting cases. To facilitate this presentation, a number of ground rules are assumed for the ensuing analysis. Most of these assumptions have to do with sizing of the various spacecraft required to perform the sample return. These dry mass requirements are taken either directly from current design points or from scaling laws based upon them. Sample size, one of the main drivers of required mass sizing, is fixed at one kilogram.

Retro propulsion is achieved with solid rocket motors, with a few exceptions. An earth-storable bipropellant (space-storable is optional) is used for Mars orbit insertions. Midcourse corrections and orbit trims are provided by monopropellant liquid rockets when a bipropellant stage is not available to do this job.

Performance results for the Mars Surface Sample Return are given in terms of available mass margins at various points of interest in the mission sequence. Minimum required mass is derived by starting with earth return systems and "backing up" through the mission sequence, adding fixed masses, retro stages, and applying impulses as called for by the options selected. From this process the required mass at launch is obtained, whether for one or two vehicles. The sequence is then reversed: beginning with the full available launch vehicle capability, impulses and scaling data are applied in the "forward" sense - through Mars arrival, descent to the surface, sample acquisition, ascent, rendezvous, departure and return to earth. The differences (margins) between available and required mass are displayed to characterize the mission as to degree of difficulty, potential mass increase, etc. Any margin which appears (e.g., at earth launch) may be propagated forward in the mission sequence to produce margins at other subsequent points.

Table 7 shows an example of the dual launch output format. One such table appears in the handbook for each option considered. Enough descriptive information is given about the mission to detail not only the planning options selected, but the underlying interplanetary trajectories as well. Note that the two launches of the dual launch mission are separated by function. One vehicle is comprised of Mars landing and ascent modules. The other is responsible for interplanetary transfer and Earth return of the sample. The mass summary at the bottom of Table 7 shows three figures at each of four critical points for each launch vehicle. The three are required mass, available mass, and the margin.

The single-launch tables present essentially the same information, but must take into account the fact that many of these

1984

TABLE 7

1984

MARS SURFACE SAMPLE RETURN

MASS PERFORMANCE SUMMARY

DUAL LAUNCH	MARS LANDER/MAV	MARS ORBITER/ERV
LAUNCH OPPORTUNITY	1984	1984
LAUNCH VEHICLE	SHUTTLE/IUS(III)	SHUTTLE/IUS(III)
MISSION OPTION	MARS ORBIT CAPTURE	EARTH ORBIT CAPTURE
LAUNCH WINDOW	20 DAYS	20 DAYS
MARS RETRO STAGE	EARTH-STORABLE(ISP=300)	EARTH-STORABLE(ISP=300)
SAMPLE SIZE	- -	1 KG

MISSION DESCRIPTION

VEHICLE	EVENT	DATE	MANEUVER	FLIGHT TIME
LNDR/MAV	EARTH LAUNCH	26 DEC 1983	C3 = 11.882	EARTH-MARS LEG 281 DAYS (TYPE II)
	MARS ARRIVAL	2 OCT 1984	DV = 1.513	
ORB/ERV	EARTH LAUNCH	26 DEC 1983	C3 = 11.882	EARTH-MARS LEG 281 (TYPE II)
	MARS ARRIVAL	2 OCT 1984	DV = 1.513	
ERV	MARS DEPART	3 MAR 1986	DV = 0.732	MARS-EARTH LEG 213 (TYPE I)
	EARTH ARRIVAL	2 OCT 1986	DV = 2.337	
TOTAL MISSION				1011 DAY 2.8 YRS

CUMULATIVE MASS SUMMARY (KG)

		REQUIRED MASS	MARGIN	AVAILABLE MASS
ORB/ERV	EARTH ENTRY.....	52	154	206
	EARTH RETURN VEHICLE	294	208	502
	MARS ORBITED MASS...	2141	514	2655
	EARTH LAUNCH.....	4128	991	5119
LNDR/MAV	MARS ASCENT.....	493	355	848
	LANDED MASS.....	769	435	1204
	MARS ENTRY.....	1206	555	1761
	EARTH LAUNCH.....	3967	1152	5119

missions are marginal performers. Therefore, the steps taken to produce (if possible) a margin at earth launch are shown in the table. Referring to the example in Table 8, the first try, with earth orbit capture and earth-storable retro at Mars, turns out to require more mass than is available. Each succeeding line shows application of one of the fallback steps and the decrease in required mass which results. If a reasonable launch margin is found, the launch window extent is expanded to a maximum of twenty days. Subsequent sections of Table 8 describe the mission and present a mass summary, showing application of available margin at three points in the mission sequence.

A brief summary of results generated for both single and dual-launch missions throughout the nine opportunities is shown in Table 9. Two additional variations on the baseline mission are included in the handbook: these achieve better mass performance by constraining the mission to space-storable retro and direct entry at Mars.

Yet another revision to the MSSR section is planned for the near term: sample sizes greater than one kilogram will be allowed. This change will necessitate substantial rescaling of field dry masses and redefinition of study groundrules to properly encompass a new range of possible missions.

MARS SURFACE SAMPLE RETURN

MASS PERFORMANCE SUMMARY

MISSION OPTION DIRECT ENTRY/MOR

LAUNCH VEHICLE SHUTTLE/IUS (III)

SAMPLE SIZE 1 KG

MASS MARGIN AVAILABLE AT EARTH LAUNCH

EARTH ENTRY OPTION	MARS RETRO SYSTEM	MARS ENTRY ORBIT	WINDOW EXTENT	LAUNCH MASS (KG)		
				REQ.	AVAIL.	MARGIN
ORBIT	EARTH-STORABLE	N/A	0 DAYS	6025	5229	-797
ORBIT	SPACE-STORABLE	N/A	0 DAYS	5141	5229	88
DIRECT	SPACE-STORABLE	N/A	0 DAYS	5024	5229	205
DIRECT	SPACE-STORABLE	N/A	10 DAYS	5128	5179	51 ***

*** - THIS CASE IS DETAILED BELOW

MISSION DESCRIPTION

EVENT	OPTION	DATE	TYPE	MANEUVER	FLIGHT TIME
EARTH LAUNCH	31 DEC 1983	II	C3 = 11.270	EARTH-MARS LEG 281 DAYS
MARS ARRIVAL	DIRECT	7 OCT 1984		VE = 6.108	MARS STOPOVER 493
MARS DEPARTURE	MOR	17 FEB 1986	I	DV = 0.710	MARS-EARTH LEG 213
EARTH ARRIVAL	DIRECT	18 SEP 1986		VE = 11.656	-----
					TOTAL MISSION 987 DAY : 2.7 YRS

CUMULATIVE MASS SUMMARY (KG)

	MASS MARGIN APPLIED TO		
	EARTH LAUNCH	MARS LANDER	ERV
EARTH ENTRY.....	30	30	30
EARTH RETURN VEHICLE	263	263	276 ←
MARS ASCENT.....	493	493	493
MARS LANDER.....	769	791 ←	769
MARS ENTRY.....	1223	1253	1223
EARTH LAUNCH.....	5179 ←	5179	5179
AVAILABLE MARGIN....	51 ←	23 ←	13 ←

Table 9

MSSR MISSION PERFORMANCE SUMMARY

MISSION DESCRIPTION	LAUNCH OPPORTUNITY									
	81	84	86	88	90	92	94	96	99	
● DUAL LAUNCH										
● Mars Orbit Capture										
○ IUS (3)	X	X	X	X	X	X	X	X	X	BASELINE
○ IUS (2/3)*	X	-	-	X	-	X	X	X	-	
○ IUS (2)	X	-	-	X	-	X	X	X	-	
● Mars Direct Entry										
○ IUS (3)	X	X	X	X	X	X	X	X	X	X = Mission
○ IUS (2/3)*	X	X	X	X	X	X	X	X	X	- = No Mission
○ IUS (2)	X	-	-	X	-	X	X	X	-	
● SINGLE LAUNCH										
○ O/MOR, IUS(3)	X	-	-	X	X	X	X	X	X	
○ D/MOR, IUS(3)	X	X	-	X	X	X	X	X	-	
○ O/MOR, Tug(R)	N/A	N/A	X	X	X	X	X	X	X	
○ D/MOR, Tug(R)	N/A	N/A	X	X	X	X	X	X	X	

* - Lander/MAV uses IUS (2), Orbiter/ERV uses IUS (3)

2.4 Penetrator Advanced Studies (1542 Man-Hours)

Advanced studies of planetary penetrators have been conducted by SAI for the past three years. These studies have focused on defining concepts and solving related problems of penetrators applied to in situ surface exploration of solar system objects. The following three subtasks were addressed in this year's studies:

Subtask 1: Lunar Penetrators Concept Study

Subtask 2: Galilean Satellite Penetrator Experiments

Subtask 3: Ad Hoc Surface Penetrator Science Committee

Earlier studies have analyzed Mars penetrator mission concepts, deployment/navigation capabilities for airless body penetrators, and penetrator missions to Mercury and the Galilean satellites. Subtask results for this year's work are briefly summarized in the next three subsections.

2.4.1 Lunar Penetrators Concept Study

The purpose of this study was to investigate the feasibility of continued exploration of the lunar surface with penetrators. Lunar penetrators have been suggested as a means for constructing a comprehensive base of in situ geophysical and geochemical information supportive of future lunar mission planning. Because we already have returned samples from several lunar sites, and considerable interest exists in performing a lunar polar orbiter mission with similar objectives, it was important that a relatively simple concept be devised for penetrators which would retain their cost-effectiveness. Also, a large variety of sites should be accessible to the penetrators in order to justify their potential contribution to our understanding of the lunar surface. Finally, a surface lifetime objective of at least one year would be highly desirable to guarantee useful seismic results.

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The concept selected for analysis in this study seeks to preserve these three criteria, i.e., low-cost, good accessibility, and acceptable lifetime. Briefly, the penetrators would be self-deployed, intended to be carried into low-earth orbit as piggy-back payloads in the Shuttle cargo bay. Each launched package would consist of a penetrator, two solid-motor stages, and a small cruise control module. The first solid motor would inject the package on a translunar trajectory. The cruise control module would provide attitude stability, guidance, and navigation during the translunar flight. It would also mark the ignition altitude for the second solid motor which would perform the lunar braking maneuver. Immediately after burn-out, the cruise control module would pitch the penetrator over to a near-zero angle of attack permitting it to free-fall to the surface. The velocity controlled impacts would result in penetration of the forebody 1-15 meters deep depending upon soil composition. The aft-body of the penetrator would be brought to rest before it became submerged thus permitting it to serve as a communication base with the earth. The separated fore and aft bodies remain connected by an uncoiled umbilical which passes electrical power and data between the two parts.

Accessibility over the lunar surface must be restricted to the front side of the moon since communications are performed directly between the penetrator aft-body and the earth. An acceptable impact zone is further constrained by a limiting flight path angle at retro ignition and the impact site earth elevation angle. These have been conservatively set at 60° and 30° , respectively. The boundary of site accessibility (for a 72-hour translunar trajectory) on the front side of the moon with these constraints is illustrated in Figure 3. Obviously, the low-cost motivated self-deployment concept does compromise site accessibility. However, even the casual observer will recognize that a wide variety of lunar features within both mare and highlands regions reside inside the impact zone which covers approximately 25% of the lunar surface.

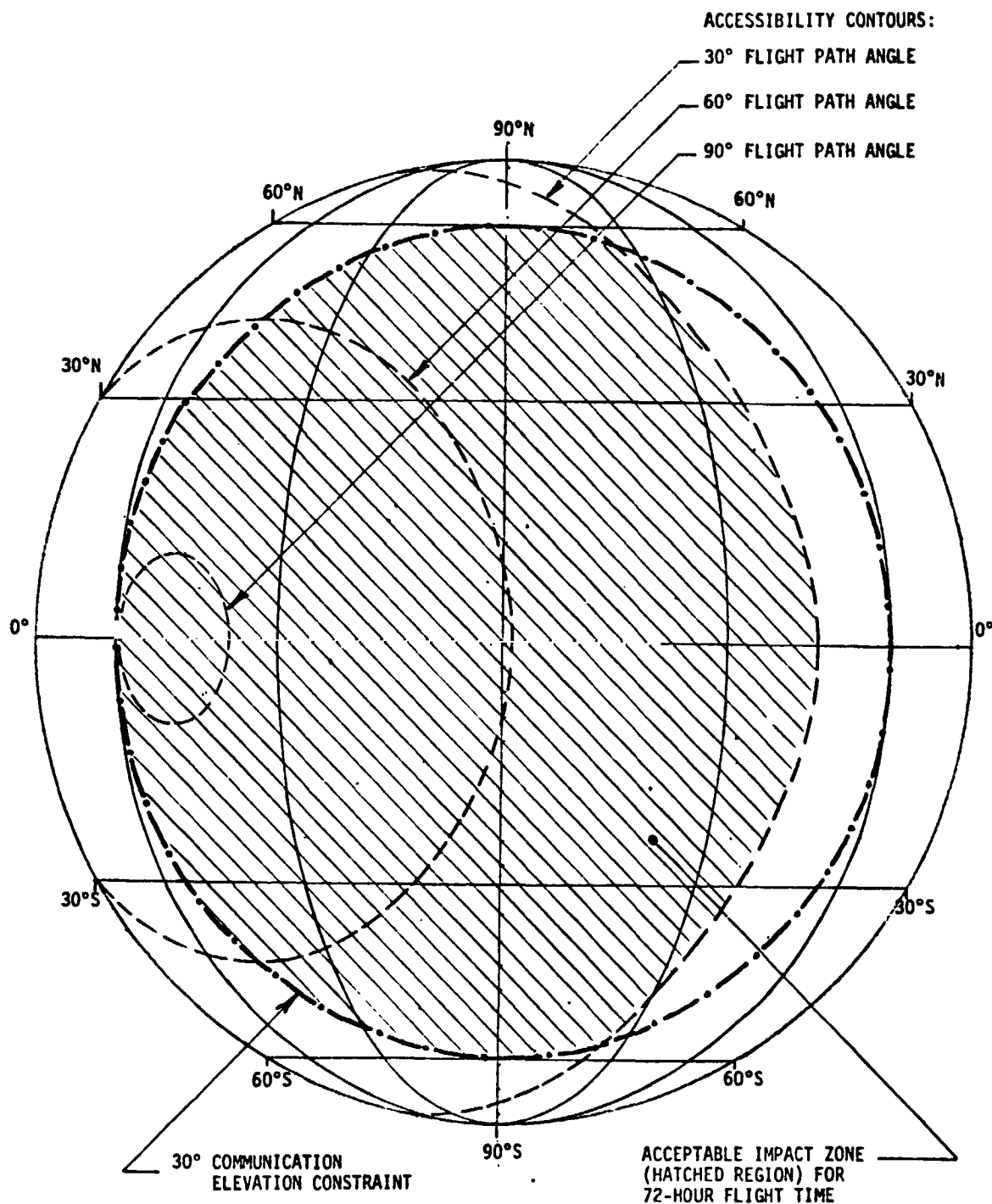


FIGURE 3. SELF-DEPLOYED LUNAR PENETRATOR IMPACT ZONE (72-HOUR FLIGHT TIME)

A candidate science payload, used in this study for analyzing system support requirements, is summarized in Table 10. Its apparent emphasis on lunar surface, subsurface and interior geophysics and geochemistry is not unlike previously suggested instruments for a Mars penetrator mission. A total of seven experiments (including aft-body panoramic imagery) and a soil sampler are included in this payload list. Only the seismometry, magnetometry, and heat flow experiments would be operated continuously over the surface life of the penetrator. The remaining experiments would be completed within the first two weeks after penetration. Typical instrument specifications and capabilities shown in the table have been taken from data developed* for a Mars penetrator. The total science payload mass is 3.8 kg, requiring about 100 mw of continuous power plus battery-supplied peak powers of up to 5w. Payload data generation is limited by storage and power requirements to not more than 1.5×10^6 bits during any 24-hour period.

Analysis of penetrator system requirements quickly isolated thermal control as the critical design issue. The penetrator design lifetime of at least one year combined with power requirements for continuously operating experiments and the central processor (computer) lead to an RTG requirement. The thermal output of the RTG (only 4% efficient) combined in turn with the very low lunar soil thermal conductivities expected even at 10m depth ($k \approx 1 \times 10^{-4}$ watt/cm²K) can lead to steady-state penetrator temperatures in excess of 400°K. A detailed assessment of instrument operating requirements and data communication loads was performed in order to determine minimum viable power requirements. The resulting power budget was split between a primary battery and RTG unit with the battery carrying as much of the power load as possible (230 watt-hr rating). The resulting RTG requirement was rated

*Mars Surface Penetrator-System Description", MSWG-84 Report, Ames Research Center, May 1977.

Table 10
CANDIDATE LUNAR SCIENCE PAYLOAD REQUIREMENTS

LOCATION	EXPERIMENT	INSTRUMENT CAPABILITY	INSTRUMENT SPECIFICATIONS				
			WT (gm)	VOL (cc)	PWR (mw) ^a	DATA (bpd)	OPER. DAYS
FOREBODY	SEISMOMETRY	3-AXIS, 0.5-5 Hz@10 ⁻⁸ G	600	400	9/90	100K-1400K	CONT.
	α, p . BK SCTR W/X-RAY	<1%(C-Fe), PPM(K-U)	400	300	0/100	6.5K	4
	γ-RAY SPECTROMETRY	PPM(H, K, Th, U)	1300	350	0/300	6.5K	1
	SOIL SAMPLING	1-4cm DEPTH	450	400	0/5000	NA	4
	IMPACT DECELERATION	5-300, μsec @ 2x10 ⁴ Hz	25	30	0/30	60K	1
BOREHOLE	HEAT FLOW	200-300°K @ .05°K	65	50	1/30	3K-170K	CONT.
AFTBODY	MAGNETOMETRY	0-100 γ @.05 γ	400	300	7/70	100K	CONT.
	IMAGERY	360°x90° SCANE@ 0.1 °FOV	250	170	0/900	1300K/PIC	>5
CONTINGENCY (@10%)			350	200	10/NA	NA	NA
TOTAL PAYLOAD			3840	2200	108/NA	<1500K	CONT.

a. Power Data: Continuous/Peak

at 4 watts thermal. With this heat source in the penetrator, it is expected to reach a temperature 200°K above ambient, virtually independent of its initial implantation temperature. For lunar subsurface temperatures in the range of 220-260°K (the higher value occurring through the lunar equatorial region) temperatures in the penetrator could reach 460°K. To bring the value down to a more desirable engineering/materials upper limit of 350°K would require finding lunar subsoils with conductivities of at least 2.2×10^{-4} watts/cm°K. Revised Apollo heat flow data and supporting laboratory experiments now lead lunar geologists to believe that maximum values of lunar subsoil conductivity won't exceed 1.5×10^{-4} watts/cm°K. It should be added that those results presume the inclusion of a heat pipe in the penetrator in order to enhance its heat loss capability. Hence, an inescapable conclusion of the systems analysis is the fact that a long-lived lunar penetrator must also be a high temperature device.

The impulse requirements for self-deployed lunar penetrator missions are summarized in Figure 4. Total impulse along with its components of translunar insertion ΔV and lunar retro ΔV are plotted against translunar flight time with curves for the moon at periapee and at apoapse. From these results it can be seen that little benefit is derived from extending flight time beyond three days (72 hours). Conversely, ΔV requirements begin to rise sharply for flight times less than one and a half days (36 hours). The position of the moon in its orbit has little effect on impulse requirements (<5% variation).

In order to complete the system analysis with an assessment of propulsion requirements, the lower-energy 72-hour flight time was selected. Translunar injection and lunar retro impulses of 3250m/sec and 2670 m/sec, respectively, were assumed from Figure 4. A midcourse navigation allowance of 100m/sec was also assumed for the 72-hour transfer. This value is large compared to traditional allowances due to the rather crude injection accuracy of the translunar injection motor. The cruise control module, which controls the penetration's

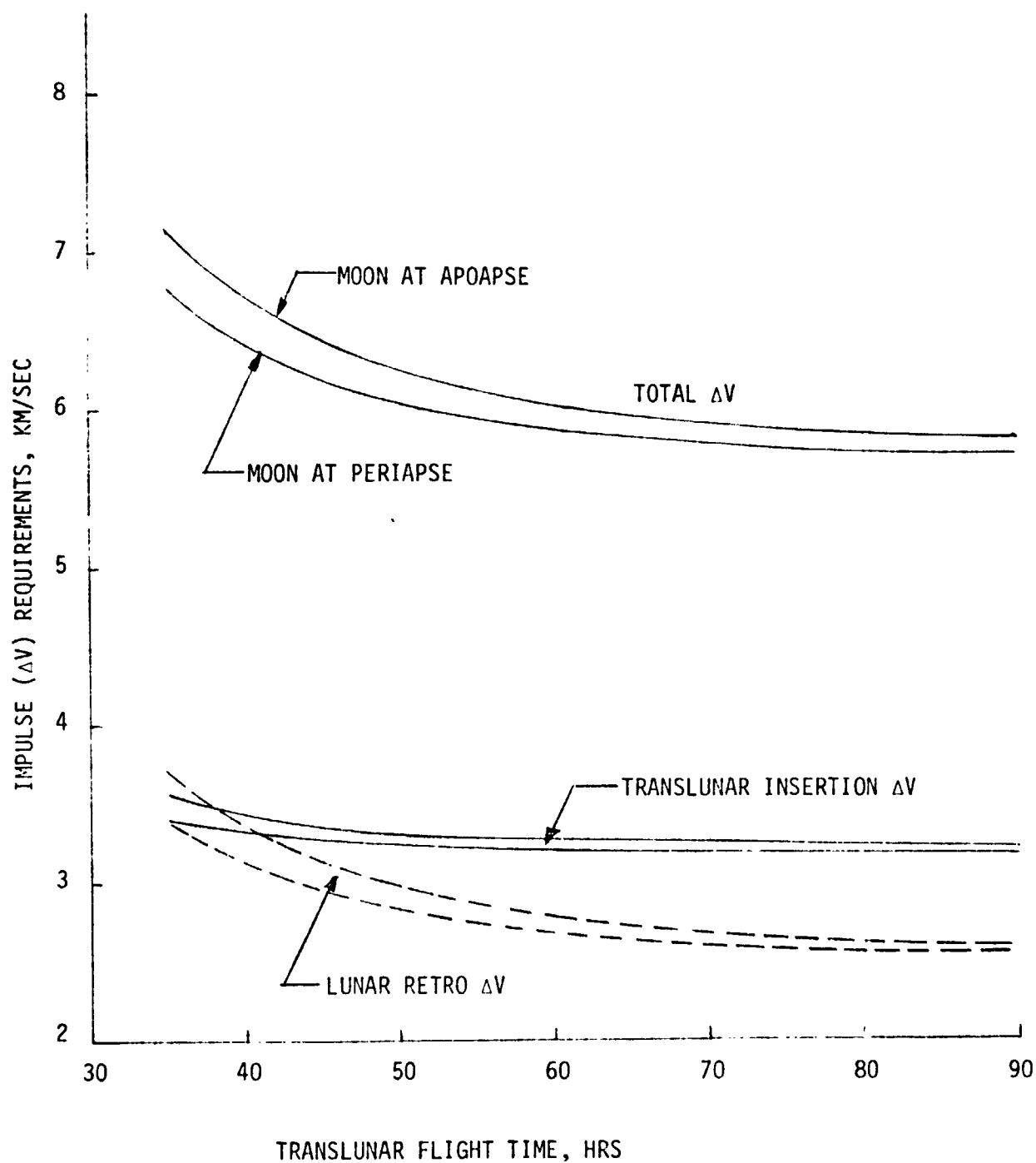


FIGURE 4. IMPULSE REQUIREMENTS FOR LUNAR PENETRATOR MISSIONS

stability during cruise uses a hot-gas system for performing the midcourse maneuvers as well as attitude control. A mass summary of the complete penetrator package is given in Table 11. The penetrator itself weighs 40 kg. Along with the cruise control module weighing 35 kg, it forms a 75 kg payload which must be transported from low earth orbit to a terminal velocity of 150m/sec at the moon. The solid motor mass requirements to do this job, plus suitable inter-stage adapter and contingency allowance increase the package to a total mass of 950 kg. This would be the mass required for each penetrator package carried in the Shuttle cargo bay. It is a little more than 3% of the Shuttle payload capacity, certainly a reasonable mass level for a piggy-back payload.

An error analysis of the self-deployed lunar penetrator impact conditions was also conducted in this study. Open-loop impact error results are presented in Table 12 for both 36-hour and 72-hour trans-lunar flight times. Data are presented for three retro flight path angles. Desired impact conditions are 150 miles at zero impact angle (angle-of-attack) with zero miss. Both the impact speed and miss errors are tolerable but the angle-of-attack (AOA) errors are much too large. Values of less than 10° AOA even in soft soils are required for successful penetration. Adding accelerometer measurements to the lunar braking maneuver significantly decreases the AOA error, assuming the cruise control module is capable of reorienting the penetrator to the prescribed attitude after burnout without error (it carries gyros and attitude sensors for this and other attitude maneuvers). Residual impact AOA with accelerometer measurements is plotted in Figure 5 as a cumulative probability distribution for the worst case flight mode, i.e., 72-hour transit time and vertical approach. Even in this case there is a 90% chance of impacting with less than 6° alignment (AOA) error. Hence, accelerometer measurements are necessary and probably sufficient means for controlling impact errors.

Table 11
LUNAR PENETRATOR MASS SUMMARY

Lunar Penetrator	40 kg
Cruise Control Module (wet)	35
Lunar Braking Motor* ($\Delta V = 2670$ m/sec)	<u>147</u>
Net Injected Mass	222
Interstage Adapter (@ 5%)	11
TLI Motor* ($\Delta V = 3250$ m/sec)	673
Contingency (@ 5%)	<u>44</u>
Total Package Mass	950 kg

*Solid Rocket: $I_{sp} = 300$ sec, Mass Fraction = 0.9

Table 12
SELF-DEPLOYED LUNAR PENETRATOR OPEN LOOP IMPACT ERRORS (3σ)

Flight Time (hrs)	Retro-Path Angle (deg.)	Impact Speed (m/s)	Impact Angle (deg.)	Impact Site (km)
36	-90	16.9	21.0	8
	-60	16.9	19.0	8
	-30	16.9	17.0	7
72	-90	12.7	18.5	9
	-60	12.7	18.5	8
	-30	12.7	15.0	8

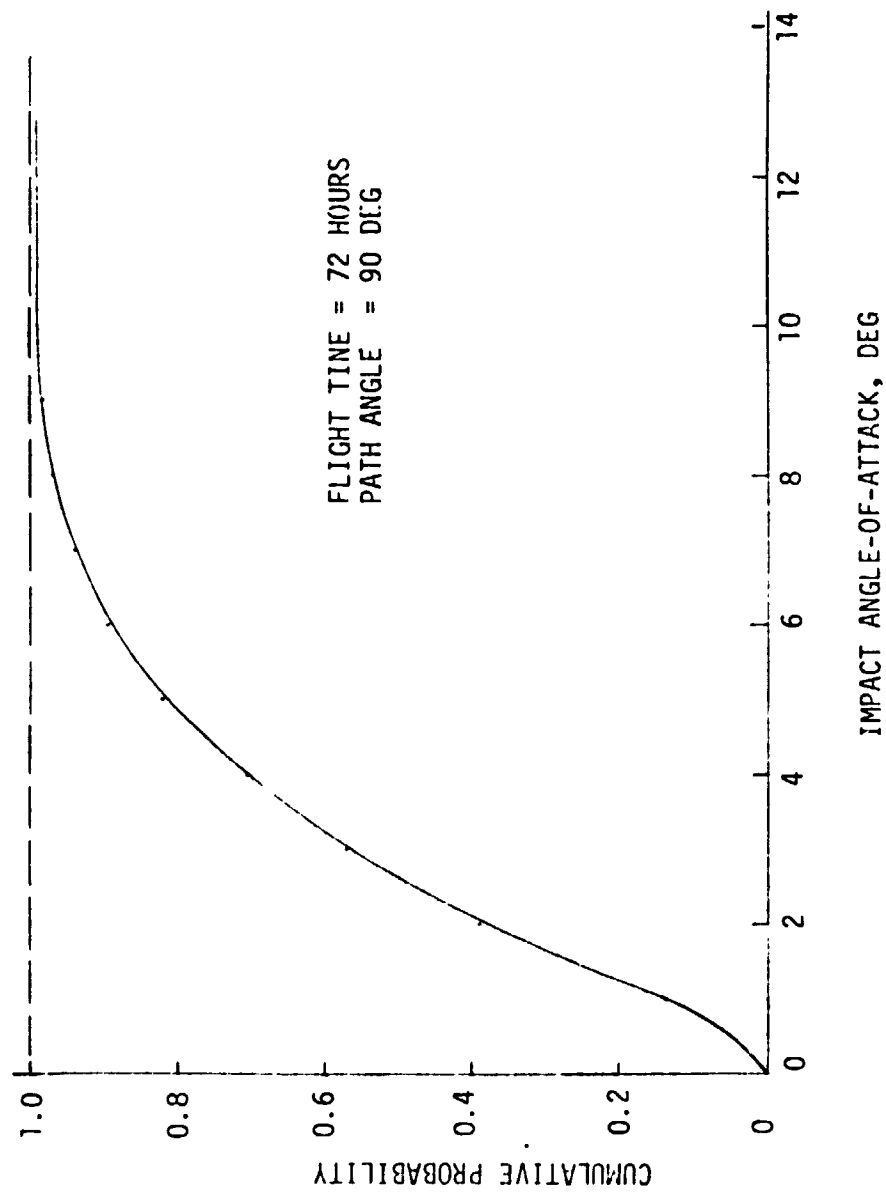


FIGURE 5. PROBABILITY DISTRIBUTION OF IMPACT ANGLE-OF-ATTACK
WITH ACCELEROMETER RETRO ERROR MEASUREMENT

In conclusion, the most serious problem uncovered by this analysis of the low-cost lunar penetrator concept is its thermal control. Steady-state temperatures of greater than 400°K certainly imply design changes. Elimination of the RTG or incorporation of high-temperature components and materials are two alternatives. Removal of the RTG seriously degrades the penetrators science capability. Incorporation of high temperature hardware probably means failure of the low-cost objective. In view of its inherently restrictive site accessibility (25% of the surface), the added problem of thermal control reduces the self-deployed lunar penetrator concept to questionable interest for future lunar exploration strategies. If a strong, but as yet unvoiced, science interest exists for the concept, a detailed systems analysis of this concept will be required to determine the exact magnitude of its thermal problems.

2.4.2 Galilean Satellite Penetrator Experiments

The objectives of this subtask were threefold: 1) to examine the feasibility of conducting geochemical penetrator experiments emplaced on the Galilean satellites; 2) to determine and compare the degradation of geochemical experiments due to Jovian trapped radiation dose effects as a function of penetrator emplacement depth at Io and Ganymede; and 3) to determine experiment degradation effects from both Jovian trapped radiation and spacecraft power sources (RTG's) for mission delivery profiles of penetrators to Io and Ganymede.

The scope of the analysis was set by the consideration of three different geochemical penetrator experiments. These were:

- 1) Alpha/Proton/X-ray Spectrometer
- 2) Neutron/Gamma-ray Spectrometer
- 3) Neutron Water Detector

The Alpha/Proton/X-ray Spectrometer performs an elemental composition inventory for small samples by detecting short range particles (alphas and protons), by detecting fluorescent x-rays excited by alpha particle bombardment, and by x-ray irradiation from suitable instrument sources. The Neutron/Gamma-ray Spectrometer performs a similar inventory of bulk samples by detecting penetrating gamma rays which occur naturally in the environment or are intensified by a neutron source carried in the instrument. The Neutron Water Detector performs a water inventory by observing the decay time and/or energy spectrum of neutrons injected into the bulk material surrounding the penetrator.

Radiation effects are important during three phases of a Galilean satellite penetrator mission: the interplanetary phase, the Jupiter orbit phase, and during operations on the satellite surface. Data on accumulated doses during the first two phases were based on previous studies updated to incorporate the latest deployment orbits for penetrators. Doses and dose rates for the surface phase were determined in this study as a function of the depth of burial of the penetrator using high energy particle transport codes for incident electrons and protons.

The characteristics of the radiation environment as a function of depth due to incident electrons was determined using the EGS/PEGS computer program for the Monte Carlo simulation of electromagnetic cascade showers. This code handles the electromagnetic interactions of electrons (negative and positive) and protons for any material up to 100 GeV. The radiation environment induced by incident protons was determined with the HETC computer program for the Monte Carlo simulation of the transport of high energy particles. The code handles the electromagnetic and nuclear interaction of protons and the neutrons, pions, muons and gamma-rays produced by high energy proton bombardment. Equivalent monoenergetic particle fluxes and radiation dose rates in Rads were determined as well as the actual particle fluxes of each species as a function of energy.

The calculations show that accumulated dose is not a problem during the landed phase of the mission, even at Io, which represents the worse case radiation effects of the Galilean satellites. However, there are still a number of constraints on instrument performance. For the Alpha/Proton/X-ray Experiment the instrument must be deeper than 20 g/cm^2 for successful operation in the alpha-mode because of electron background. The same instrument must be deeper than 50 g/cm^2 for successful operation in the proton-mode because of proton and electron background. For the Neutron Gamma Experiment, gamma ray line emission excited by trapped proton bombardment dominates other sources of line emission (radioactive elements, galactic cosmic ray bombardment, on board neutron source) at all depths less than 400 g/cm^2 and the detector must be deeper than 150 g/cm^2 to keep bremsstrahlung and electron count rates in a range acceptable for pulse height analysis. Finally, for the Neutron Moderation (water) Experiment, the instrument must be deeper than 400 g/cm^2 if the water content is low because the neutron background interferes with observations of the decay of neutron pulses injected by the instrument. Parametric calculations also performed for Europa, Ganymede and Callisto show more favorable situations at those satellites.

Radiation accumulated during the Jupiter orbital phase represents a severe constraint on penetrator experiments to the inner Galilean

satellites using presently conceived deployment profiles. Multiple, satellite assisted orbits are needed to reduce approach speeds at Europa or Io to levels compatible with reasonable deployment braking motors for penetrators. The accumulated radiation dosage during these revolutions far exceeds acceptable limits on electronic spacecraft and penetrator components. Hence, new concepts to emplace the penetrator on the surface much more rapidly are needed if penetrator experiments at the inner satellites are to be feasible. Furthermore, penetration depths of at least 3m are required at Io for satisfactory performance of all instruments as discussed above. These combined radiation effects limit currently feasible Galilean satellite penetrator missions to Ganymede and Callisto. Penetrator mission concepts for Io and Europa require further study in light of these identified radiation hazards before feasibility can be assured.

2.4.3 Ad Hoc Surface Penetrator Science Committee

The purpose of this subtask was the provision of engineering support to the Ad Hoc Surface Penetrator Science Committee organized by Ames Research Center (ARC) at the direction of NASA Headquarters. This committee was formed as part of the FY 1976 Penetrator Development Program being conducted by ARC. Its purpose was to provide assurance that maximum science potential of the surface penetrator and its science payload would be realized. To render such assurance, the Committee was to convene from time to time to study the concept, its application to planetary exploration, and the concurrent penetrator sensor development program being conducted by ARC. SAI engineering support was provided through the membership of John Niehoff, SAI Advanced Studies Leader, on the Committee. Complete committee membership was as follows:

Prof. J. Westphal	California Institute of Technology	Chairman
Dr. D. Currie	University of Maryland	Physics
Dr. J. Fruchter	Patelle Northwest Laboratories	Geochemistry
Dr. J. Head	Brown University	Geology
Dr. C. Helsley	University of Texas (Dallas)	Seismology
Dr. C. Lister	University of Washington	Geophysics
Dr. J. Tillman	University of Washington	Meteorology
Mr. J. Niehoff	Science Applications, Inc.	Engineering

The committee met three times during its tenure in the spring of 1976. The first meeting was held at Ames Research Center on January 22-23, 1976. In addition to settling various organizational and functional issues, the committee members received tutorials related to science disciplines of potential penetrator science and had an opportunity to ask many related questions of application and feasibility. Their second meeting was held April 30, 1976 in conjunction with a two-day Program Review of the ARC Development Program at Albuquerque, NM. A thorough review of instrument development, penetrator deployment, and soil contamination studies was presented to the Committee. It was at this meeting that the Committee evolved a ranking system for classifying potential penetrator science experiments. That system of experiment class definitions is as follows:

- Class 1: Essential
- Class 2: To be included, if feasible
- Class 3: Highly desirable, if feasible and provided there is no major negative impact of Class 1 and feasible Class 2 experiments
- Class 4: Secondary, use as accommodation permits, if feasible and provided there is no significant negative impact on Class 1, 2 and 3 experiments.

Using this system, the Committee also reached a preliminary classification of proposed experiments, in the context of an early Mars mission at this time. The third and final meeting of the Committee was held August 6-7, 1976 at Cal Tech to finalize our conclusions and recommendations, and incorporate these findings into a final report. Among the significant activities accomplished at this meeting was a finalized classification of proposed penetrator experiments. In the format of the class priorities outline given above those results were as follows:

- Class 1: Seismic Measurement
Imagery
- Class 2: Chemical Composition
Heat Flow
Total Water Measurement
Meteorology (Temperature, Pressure, Wind)

Class 3: Frost and Dust Detection

Organic Geochemistry (Re-evaluate after Viking GCMS
results are available)

Class 4: Ion Geochemistry

Magnetometry

Nutrient-Induced Biology

Atmospheric Relative Humidity

Soil Electrical Conductivity

It became apparent to the Committee, in the process of these classifications that a minimum Mars penetrator mission must consist of the Class 1 experiments and at least one Class 2 experiment. Detailed discussions of what Class 1 and 2 experiments should accomplish were also generated and included as part of the final report.

In addition to attendance and participation in the Committee meetings, Mr. Niehoff performed two additional activities as part of this subtask. First, he gave an interim presentation of Committee findings on May 7, 1976 to the COMPLEX at the request of chairman, Prof. Westphal. Second, he prepared a number of visual summaries of Committee results for presentation to the NASA Associate Administrator for Space Science and participated in that presentation August 12, 1976 at NASA Headquarters. Examples of these data are presented in Tables 13, 14 and 15 which summarize Mars penetrator characteristics, a 1981 Mars penetrator mission and a cost estimate of the 1981 mission based on maximum use of PVO hardware and inheritance. These data were provided in support of the Committee's recommendations to NASA. Those recommendations are presented in Table 16.

Table 13

BASELINE MARS PENETRATOR CHARACTERISTICSMASS SUMMARY

Forebody	28.7 kg
Afterbody	<u>2.3</u>
Implanted Penetrator	31.0
Decelerator	14.6
Deorbit Motor (80 m/sec)	7.0
Launch Tube	<u>7.5</u>
Total System Mass	60.1 kg

PAYLOAD CONSTRAINTS

Science/Electronics Mass	7.3 kg
Forebody Compartment Dim.	7.6 cm dia. x 102 cm long
Volume	4500 cm ³
Total Power	300 mW
Daily Energy Budget	7.2 watt-hrs
Science	3.5 watt-hrs
Data Processing, Comm.	3.7 watt-hrs
Memory Capacity	10 ⁵ to 10 ⁶ bits

INDUCED ENVIRONMENTS

Deceleration-Forebody Peak	1800 g
Afterbody Peak	18000 g
Thermal Exhaust	20 watts
Radiation	10 ⁵ neutrons/sec
Depth of Placement	1-15 m
Attitude	≤15°
Physical	Comminution and Fracture of Surrounding Material

Table 14

1981 MARS PENETRATOR MISSIONCHARACTERISTICS

Penetrators.	6 (Mars Baseline Design)
Orbiter.	PVO Modified Bus (No Science)
Launch Vehicle	Atlas/Centaur/TE-364
Orbit.	24.6-hr, Near-Polar; $H_p = 1000$ km
Launch Dates	November 18-28, 1981
Arrival Dates.	September 16-18, 1982
Completion	September 1984

MASS SUMMARY

Orbiting Bus	350 kg
Orbit Control Expendables.	25
Orbit Deployed Penetrators (4)	208
OIM Inerts	38
Hardware Contingency	<u>40</u>
Total Orbited Mass	661
Orbit Insertion Propellant	336
Approach Deployed Penetrators (2).	104
Transfer Control Expendables	<u>50</u>
Net Injected Mass	1151
L/V-S/C Adapter.	<u>24</u>
Total Launched Payload	1175 kg
Launch Vehicle Capability.	-1175 kg

Table 15

PRE-PHASE A COST ESTIMATE OF 1981 MARS PENETRATOR MISSION

	<u>FY '74 \$M</u>	<u>FY '79 \$M</u>
Program Mgt/Design	4.7	7.0
Penetrator Science	12.2	18.1
PVO* Modified Bus	46.7	69.3
6 Penetrators + 1 Spare	31.9	47.4
Mission Operations	16.3	24.2
Flight Support	3.9	5.8
Penetrator Sterilization	1.0	1.5
APA	<u>11.7</u>	<u>17.4</u>
TOTAL	128.3	190.7
<hr/>		
	Real Years Dollars (M)	215.5 (Total)
		30.4 (1st Yr)

*PVO: Pioneer Venus Orbiter

Table 16

RECOMMENDATIONS OF THE 1976 AD HOC SURFACE
PENETRATOR SCIENCE COMMITTEE

We firmly believe that penetrators represent a valuable and necessary platform for the conduct of certain essential in situ experiments in the exploration of a majority of solid solar system bodies. Therefore:

1. We recommend that, for both science and engineering reasons, the first penetrator mission undertaken be to Mars, and that this be done during the 1981 launch opportunity;
2. We understand that the scope of a 1981 Mars Penetrator mission may necessarily be dictated by a highly constrained NASA budget. We therefore recommend that a minimum viable mission must consist of at least 4 penetrators, and that each of these penetrators must carry a seismometer, an afterbody imager, and at least one of the following additional experiments:
 - a) chemical composition
 - b) total water measurement
 - c) heat flow
 - d) afterbody meteorology.

In our opinion, with reasonable effort, it will be possible to fly all of these experiments plus a few others.

2.5 Mercury Missions Transport Study (1321 man-hours)

The objective of this task was to provide a data base and comparative performance analyses of alternative flight mode options for delivering a range of payload masses to Mercury orbit. Launch opportunities over the period 1980-2000 were considered. Extensive data trades were developed for the ballistic flight mode option utilizing one or more swingbys of Venus. Advanced transport options studied include solar electric propulsion and solar sailing. Study results show the significant performance tradeoffs among such key parameters as trip time, payload mass, propulsion system mass, orbit size, launch year sensitivity and relative cost-effectiveness. Handbook-type presentation formats, particularly in the case of ballistic mode data, provide planetary program planners with an easily used source of reference information essential in the preliminary steps of mission selection and planning.

2.5.1 Ballistic Flight Mode Summary

The scope of ballistic mission data is delineated by the opportunity/configuration matrix shown in Table 17. Every case examined is characterized by launch year, number of Venus swingbys, launch vehicle upper stage and retro propulsion type. Trajectory characteristics for each opportunity are summarized in Table 18 which lists flight time, launch energy C_3 , trajectory shaping midcourse maneuvers $\Delta V_{M/C}$, hyperbolic excess approach speed V_{HP} , and the required total (post-launch) impulse budget, ΔV_N , including navigation maneuvers. A 10-day launch window assumption is reflected in the given data. From the C_3 and ΔV_N columns it may be inferred that the 1986 V(3), 1988 V(2), 1994 and 1996 launches are among the better opportunities through the end of the century.

Figure 6 shows an example of a computer-generated performance graph for the 1988 Mercury orbiter opportunity. A space-storable retro is utilized for all midcourse and orbit insertion maneuvers, and appropriate finite thrust penalties are accounted for in the values of net orbited payload - the latter being defined as useful spacecraft

TABLE 17

MERCURY ORBITER TRANSPORT STUDY

Ballistic Opportunity/Configuration Matrix

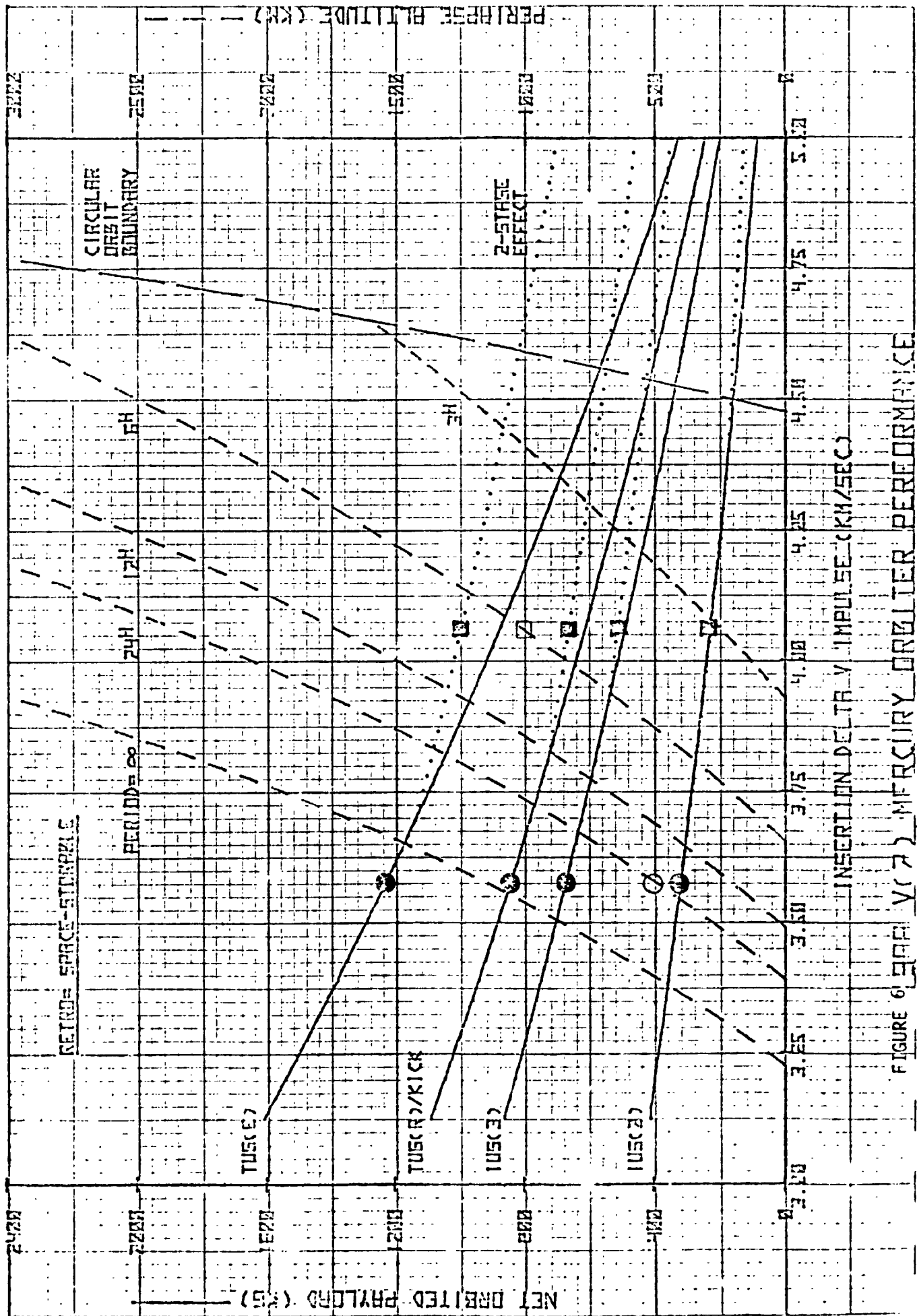
LAUNCH YEAR	VENUS SWINGBYS				LAUNCH VEHICLE OPTIONS SHUTTLE UPPER STAGE				RETRO PROPULSION OPTIONS		
	V(1)	V(2)	V(3)	V(4)	IUS(II)	IUS(III)	TUG(R)/KICK	TUG(E)	E/S	S/S	SOLID/MONO
1980	*		*		*	*			*		*
1981		*			*	*			*		*
1983		*	*		*	*			*		*
1985	*				*	*	*	*	*	*	*
1986		*	*		*	*	*	*	*	*	*
1988		*			*	*	*	*	*	*	*
1989		*			*	*	*	*	*	*	*
1991		*			*	*	*	*	*	*	*
1994		*			*	*	*	*	*	*	*
1996			*		*	*	*	*	*	*	*
1999				*	*	*	*	*	*	*	*

TABLE 18
BALLISTIC MODE CHARACTERISTICS SUMMARY

10-Day Launch Window

LAUNCH YEAR	TRANSFER TYPE	FLIGHT TIME (DAYS)	C_3 (km/sec) ²	$\Delta V_{M/C}$ (km/sec)	V_{HP} (km/sec)	ΔV_N^* (km/sec)
1980	V(3)	1126	30.90	0	6.070	0.263
1980	V(1)	657	34.20	0.100	6.650	0.196
1981-a	V(2)	1067	32.80	0.357	5.619	0.519
1981-b	V(2)	422	45.41	0.069	7.130	0.239
1983	V(2)	989	17.45	0.610	5.792	0.771
1983	V(3)	953	25.25	0	6.517	0.263
1985	V(1)	420	49.60	0.400	6.265	0.528
1986	V(2)	911	24.44	1.564	5.809	1.725
1986	V(3)	1247	19.17	0.054	5.645	0.291
1988-a	V(2)	741	25.80	0.200	6.160	0.364
1988-b	V(2)	621	28.05	0.574	5.995	0.735
1989	V(2)	792	43.25	0.230	5.858	0.393
1991	V(2)	1019	25.80	0	6.585	0.199
1994	V(2)	877	19.38	0.130	5.753	0.296
1996	V(3)	782	23.00	0	6.200	0.263
1999	V(4)	1177	26.35	0	6.100	0.323

*Values include $\Delta V_{M/C}$



mass in orbit exclusive of all propulsion system mass. Payload performance curves (solid lines) are superimposed over curves representing the orbital parameters, orbit period and periapse altitude. The latter curves are bounded on the right by the circular orbit limit, and on the left by the parabolic orbit limit. The dotted payload curve shows the performance gain available by employing a two-stage retro for orbit insertion.

Some simple examples illustrate how these curves can be used. Consider first, a case where a 24-hour orbit at 500 km periapse altitude is desired. To determine the payload which could be delivered by each of the four launch vehicles, first bring a horizontal from the 500 km mark on the right-hand axis to its intersection with the dashed curve representing a 24-hour period. This intersection is indicated in Figure 6 by an empty circle. A vertical drawn through this point intersects the four payload curves at the points indicated by the filled circles. The net orbited payload for each launch vehicle can then be read off the left-hand axis.

As another example, suppose the desired orbit has a period of 6 hours and a periapse altitude of 1000 km. The intersection of the 6-hour curve and the 1000 km horizontal is identified in Figure 6 by an empty square. A vertical drawn at this point now intersects seven payload curves. The intersection with the four solid curves gives, as before, the net orbited payload delivered by the four different launch vehicles using single stage retros. The intersection with the three dotted curves, as represented by the filled squares, determines the payload delivered by the indicated launch vehicles and two-stage retros. The fact that the dotted and solid curves coincide at the point of intersection for IUS (II), signifies that a two-stage retro would offer no performance advantage over a single stage unit for that particular launch vehicle and orbit.

2.5.2 Low-Thrust Flight Mode Summary

SEP payload performance for achieving close circular orbit at Mercury is shown as a function of flight time in Figure 7. Payload delivery in the desirable range (500-1000 kg) generally requires a transfer in the 2.5-3.5 revolution class with flight times between

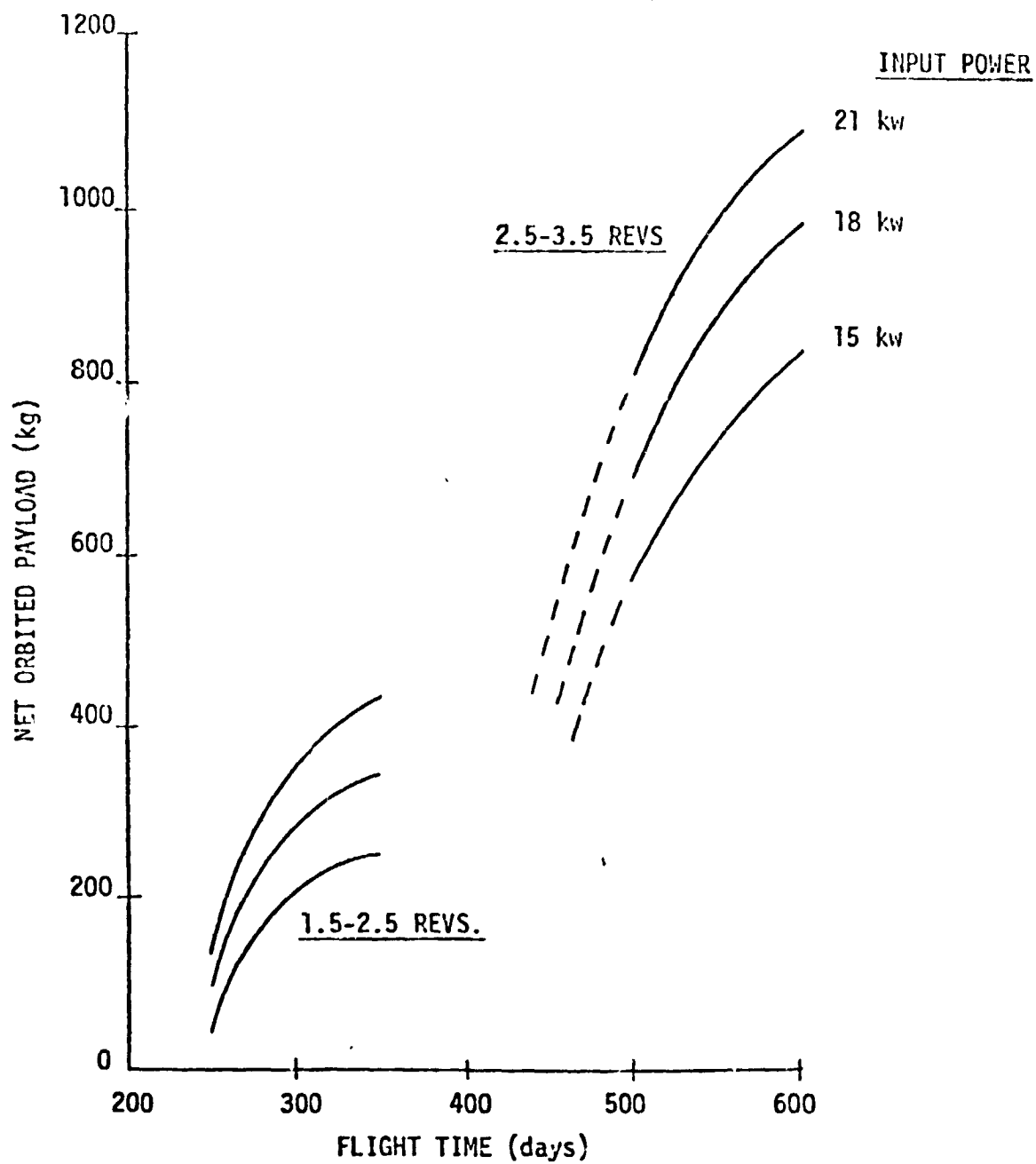


FIGURE 7 SEP PERFORMANCE FOR MERCURY ORBITER MISSION

- 1984 LAUNCH
- SHUTTLE/IUS(III)
- 500 km ALT. CIRCULAR ORBIT
- EARTH-STORABLE RETRO

500 and 600 days. Although a 15 kw system is probably adequate for some orbiter mission concepts, a higher power level of 18 or 21 kw offers significant payload gain which may be necessary for surface exploration missions. Performance sensitivity to launch year is shown in Figure 8. This effect is generally attenuated in comparison with Venus swingby ballistic transfers. The cyclical variation is about $\pm 8\%$ from the average.

Solar sail performance is presented in Figure 9 as curves of net orbited payload versus flight time for several values of sail size. The Shuttle/IUS (II) launch vehicle is capable of delivering a payload of 600 to 900 kg when a square sail size of 400 meters or less is used. Considerably greater payload performance (up to 2000 kg) is offered by the three-stage IUS. Relatively short trip times of about one year are possible with the sail transport mode.

2.5.3 Payload/Cost Comparisons

Figure 10 compares payload/flight time performance of the three flight modes for achieving a 500 km circular orbit at Mercury. Use of the Shuttle/IUS (III) launch vehicle is assumed. The five sample ballistic opportunities shown on the graph span the range of ballistic mission performance, i.e., flight times between 750 and 1250 days and orbited payloads between 250 and 650 kg. Retro system capability, in order of increasing performance, is earth-storable, solid/mono-propellant and space-storable. Solar electric propulsion offers a considerable performance improvement in terms of reduced flight time (500-600 days) and payload increases to the level 500-1000 kg. This potential of low-thrust transport is further enhanced by the solar sailing concept which could deliver sufficient payload for multiple surface lander deployment missions.

Estimates were made for the recurring cost of the transport vehicle (SEP or solar sail) and the total costs of the chemical retro systems used for each mode of flight. Figure 11 shows a comparison of the three flight modes in terms of a specific cost index, i.e., propulsion system cost per kilogram of delivered payload plotted as a function of flight time. Since low specific cost and short flight times are most desirable, it is seen that solar sailing provides the

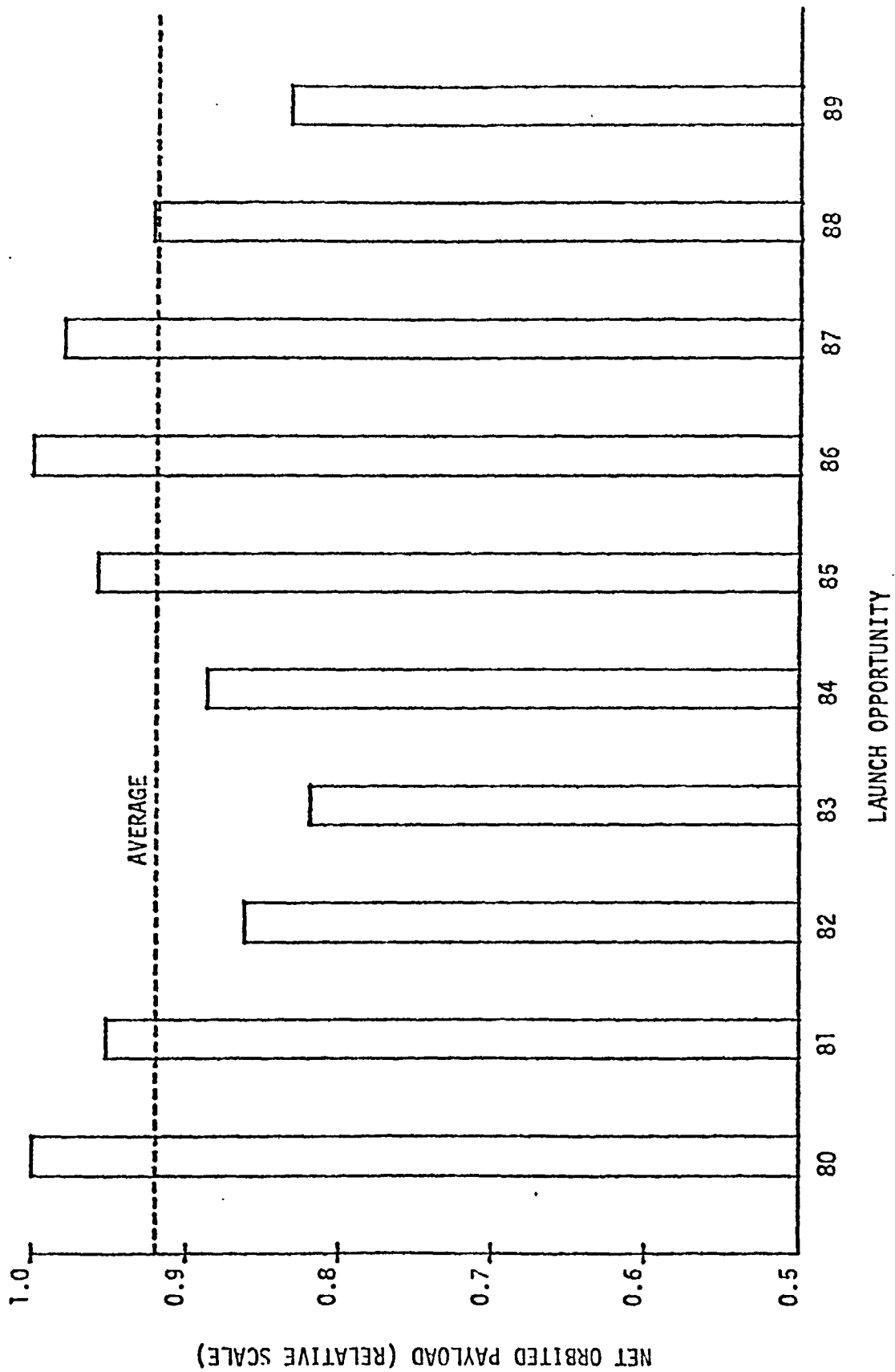


FIGURE 8 LAUNCH OPPORTUNITY EFFECT FOR MERCURY ORBITERS - SEP TRANSPORT MODE

LAUNCH ENERGY C_3

25 (km/s)^2

ORBIT SIZE

500 km CIRCULAR

EARTH-STORABLE RETRO

300 sec ISP

SAIL MATERIAL DENSITY

4 g/m^2

STRUCTURE MASS

$0.2 M_{\text{SAIL}} + 50 \text{ kg}$

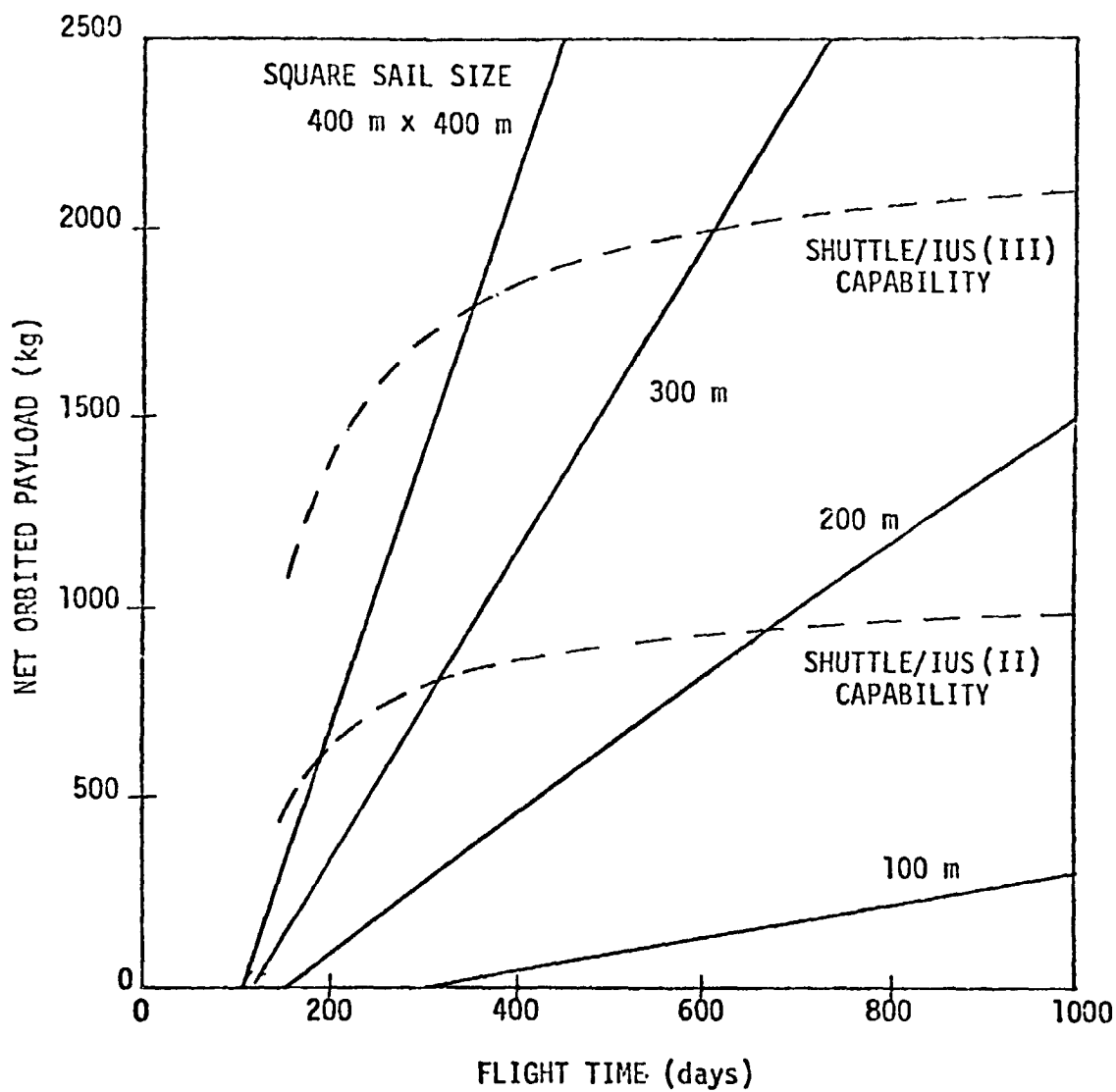


FIGURE 9 SOLAR SAILING PERFORMANCE FOR MERCURY ORBITER MISSION

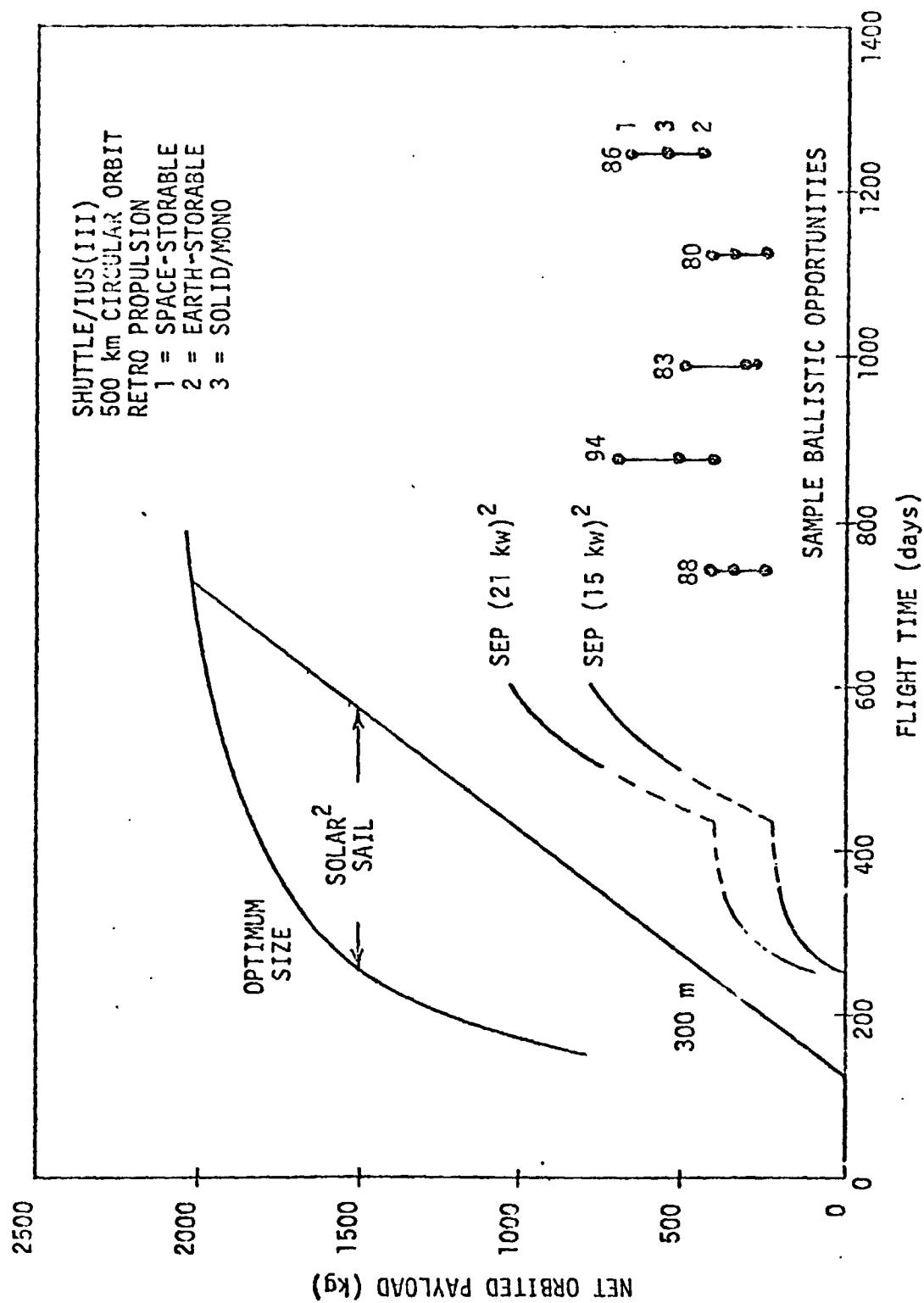


FIGURE 10 PAYLOAD PERFORMANCE COMPARISON FOR MERCURY ORBITER MISSIONS

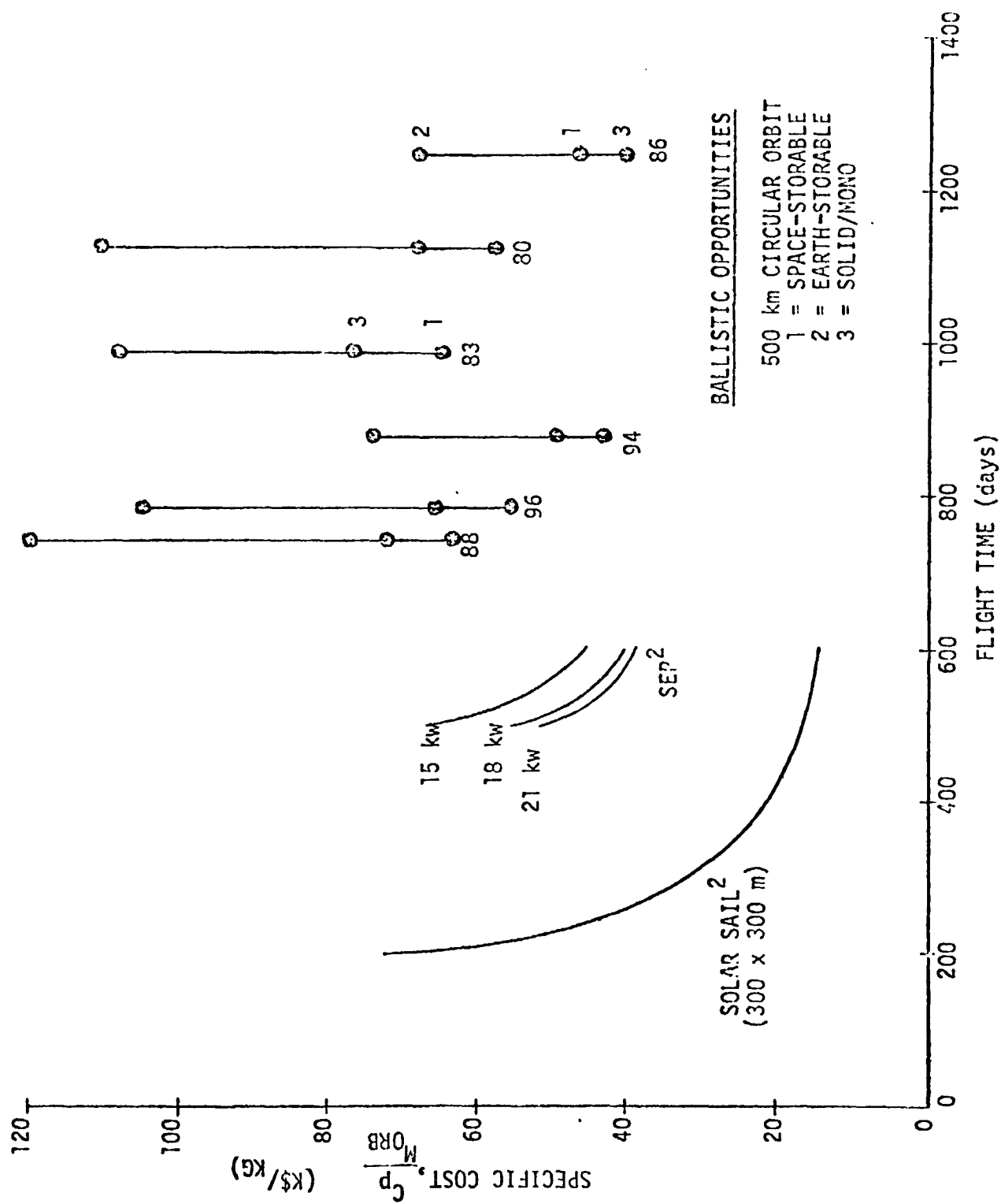


FIGURE 11 PROPULSION SYSTEM SPECIFIC COST COMPARISON FOR MERCURY ORBITER MISSIONS

best performance, followed by SEP and then ballistic mode transport. In the ballistic case, the most cost-effective retropropulsion is generally the combined solid/mono system, followed closely by space-storable, with earth-storable systems being least cost-effective.

In making the above comparison between SEP and solar sailing, the basic assumption used was a SEP recurring cost of \$20M-\$24M and a considerably lower sail recurring cost of \$6M (FY 1977 base period). Furthermore, the payload performance stated for SEP was based on current technology parameters. Since these assumptions are certainly subject to question, a sensitivity analysis was performed and the comparative results are shown in Figure 12. One may conclude, for example, that a SEP vehicle of advanced design is more nearly comparable with a solar sail vehicle in terms of cost-effectiveness.

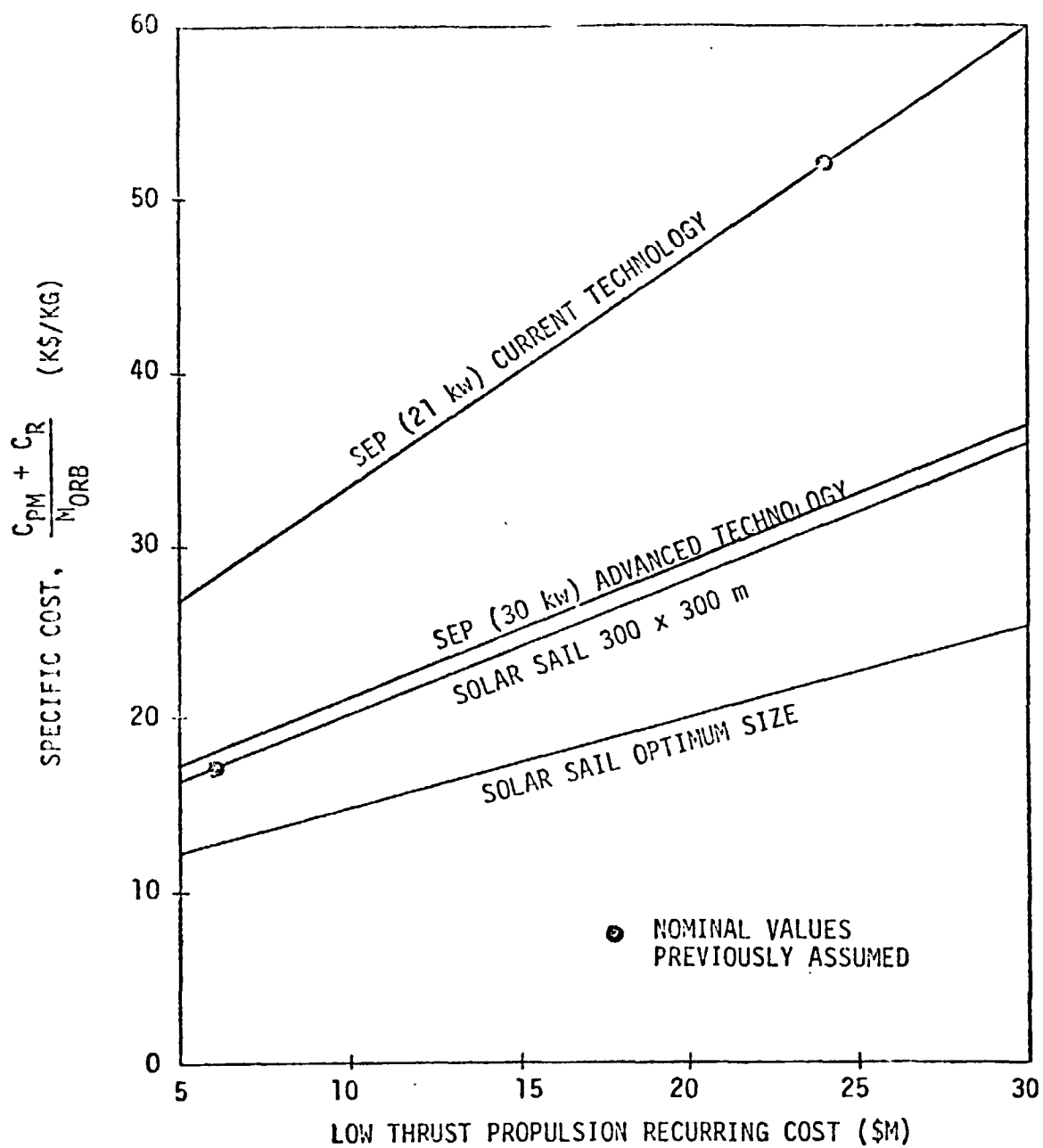


FIGURE 12

PARAMETRIC COST COMPARISON OF SEP AND
SOLAR SAIL FOR MERCURY ORBITER MISSION
TF = 500 days; 500 km CIRCULAR ORBIT
EARTH-STORABLE RETRO

2.6 SSEP/SAIL Discriminator Definition (325 man-hours)

Late in the 1976-7 Advanced Studies contract period NASA Headquarters undertook a study initiative to analyze the merits of two alternative low thrust propulsion systems for future earth-orbital and interplanetary transportation. These systems are Super Solar Electric Propulsion (SSEP) and Solar Sail (SAIL). The analysis plan for the 1977 fiscal year was to conduct studies of the technology base, design requirements, development plans, mission applications and cost of these competing future transportation system. Then in the Summer of 1977, a decision would be reached, based on assessment of these study results, as to which system NASA would pursue with the first application intended to be a Halley Rendezvous mission launched in 1982.

The purpose of this task was to prepare mission specific performance criteria with which both SSEP and SAIL transport system effectiveness could be compared and assessed. In other words, criteria of discrimination (discriminators) were to be evolved which could be used in one of the study elements of the 1977 SSEP/SAIL Activity, i.e., assessment of the utility of these systems applied to planetary missions.

A key initial step in this task was the development of baseline definitions for a set of representative planetary missions. These baselines would then be used to evolve mission-specific discriminators for comparing the two systems. Six missions were defined for this purpose. They were as follows:

<u>Mission</u>	<u>Launch Period</u>
Halley Rendezvous w/Nucleus Probe	1981-82
Saturn Orbiter w/Titan Lander	1986
Mars Sample Return	1988
Mercury Orbiter w/Rough Landers	1987-89
Asteroid Survey w/Penetrators	1988-90
Comet Sample Return	1990-95

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Baseline definitions were evolved for each of these missions including a statement of objectives, required spacecraft and probe hardware, and key mission parameters. Tabular summary results of this work are presented in Tables 19-24. Additional analysis of a specific 1991 Comet Encke sample return mission was performed as part of the baseline definition of the comet sample return mission to insure low-thrust performance feasibility and the existence of an acceptable opportunity in 1990-95 time frame.

The next step in this definition task was to itemize a minimum set of mission parameters which the SSEP and SAIL Teams at JPL would have to derive from these baseline definitions as inputs to a subsequent comparison assessment. These parameters are summarized in Tables 25 and 26 for SSEP and SAIL systems respectively. The provision of these data, consistent with the baseline mission definitions would constitute the base of information against which benefits and impacts of applying either low-thrust propulsion system to the reference planetary mission set could be assessed.

The third step in the definition task was the development of specific discriminators (mission parameters) which should be assessed for each mission. This analysis was begun with the development of a substantial list of discriminators which was subsequently broken down into four categories and iterated for completeness. Those categories were: performance, science, spacecraft, and navigation. A total of 28 discriminators were evolved by this process. A brief description of each of these discriminators, presented in category groups, is given below.

A. PERFORMANCE DISCRIMINATORS

Launch Vehicle

Within the constraint of the Shuttle/IUS system there are three IUS configurations appropriate to SSEP or SAIL missions: 1) the two-stage IUS, 2) the twin-stage IUS, and 3) the three-stage IUS. For cost and operational reasons, in general, the smaller the IUS configuration is, the more attractive becomes the mission, all other things being equal.

Table 19

SSEP/SAIL BASELINE MISSION DESCRIPTION

MISSION Halley Rendezvous w/Nucleus Probes

OBJECTIVES To conduct extensive investigation of the comet during its 1986 apparition in order to determine its:

- a) physical and chemical properties
- b) dormant/active states and transitions
- c) interaction with interplanetary media

SPACECRAFT New design (probably similar to inner planet geochemical orbiter); axis-stabilized; autonomous operations req'd; $\sim 500 \text{ kg}^a$

PROBE(S) Mars penetrator design with larger tube-launcher; retro and cruise A/C capability; battery power preferred; $\sim 7 \text{ kg}^b$ ea.; 2 units desired.

MISSION PARAMETERS

LAUNCH PERIOD: 1981-2, ARRIVAL TIME: $-50(-50/+100)$ days from T_p^c

SPACECRAFT STAYTIME: $>150^d$ after T_p , PROBE(S) LIFETIME: >30 days

CIRCUMNAVIGATIONS: TBD; function of arrival date, staytime, and conditions of jettison and deployments.

STATIONKEEPING: TBD; function of arrival date, staytime, and conditions of jettison and deployments.

DEPLOYMENTS: Both penetrators deployed as soon as possible (within 20 days) after rendezvous.

COMMENTS

- a) net mass excludes all propulsion
- b) all-up mass on board spacecraft
- c) slow flyby may be a necessary fall-back option; max. Vhp of 2.5 km/sec at $T_p + 25^d$ may still permit penetrator deployment.

Table 20
SSEP/SAIL BASELINE MISSION DESCRIPTION

MISSION Saturn Orbiter w/Titan Lander

OBJECTIVES To conduct explorative investigations of the Saturnian system including fields/particles mapping, satellite and Ring studies, planetology remote sensing, and in situ experiments on Titan.

SPACECRAFT JO design; dual spin stabilized in orbit; Titan-assisted capture; net mass allowance of $\sim 500 \text{ kg}^a$.

PROBE(S) New design (MMC OAST Study); bioshielded; active deflection; parachute descent; gross mass allowance of $\sim 400 \text{ kg}^b$.

MISSION PARAMETERS

LAUNCH PERIOD: 1986, ARRIVAL TIME: < 6 years after launch

SPACECRAFT STAYTIME: > 18 mos., PROBE(S) LIFETIME: ~ 3 mos.

ORBIT PARAMETERS:

<u>DEFINITION</u>	<u>FUNCTION</u>
1) 19.5 Rs periapse, 95.7 day period	Titan-assisted capture
2) 15.9 day period, Titan sync for 3 mos.	Lander communications
3) variable	mapping/satellite enc.

DEPLOYMENTS: Titan lander deployed prior to orbit capture, 5×10^6 km from Titan; orbiter on 1000 km periapse miss approach.

COMMENTS a) net mass excludes all propulsion
b) all-up mass added to orbiter bus

Table 21
SSEP/SAIL BASELINE MISSION DESCRIPTION

MISSION Mars Surface Sample^a Return

OBJECTIVES To retrieve precollected samples of Mars assembled (by rovers from a previous mission) at a single surface site for pick-up and return to earth orbit.

SPACECRAFT Mission module integrated with transport vehicle; highly inherited new design; includes soft-dock capability; ~400 kg.

PROBE(S) 4865 kg^b; see schematic diagram on reverse side for breakdown of hardware elements.

MISSION PARAMETERS

LAUNCH PERIOD: 1988, ARRIVAL TIME: optimum for 3-year mission

SPACECRAFT STAYTIME: ≥ 90 days

ORBIT PARAMETERS:

<u>DEFINITION</u>	<u>FUNCTION</u>
1) 1000 km altitude circular (10 orb/day)	Transport Parking Orbit
2) 40x1000 km altitude (-5° entry angle)	Entry Orbit
3) 100x950 km altitude	Ascent Orbit
4) 950 km altitude circular	Phasing Orbit

DEPLOYMENTS: Lander Deployment from 1000 km circular orbit;
 $\Delta V = 210$ m/sec; solid retro braking.

RECOVERIES: Ascent Vehicle recovery at 1000 km circular; ascent payload active/transport system passive; terminal $\Delta V = 100$ m/sec.

EARTH RETURN ORBIT: 175 nm circular orbit for direct Shuttle recovery.

COMMENTS a) ≤ 50 kg sample
 b) Derived from scaling relationship used in past MSSR studies

Table 21 (Cont'd)

SCHEMATIC SUMMARY OF MARS SURFACE SAMPLE RETURN PAYLOAD REQUIREMENTS

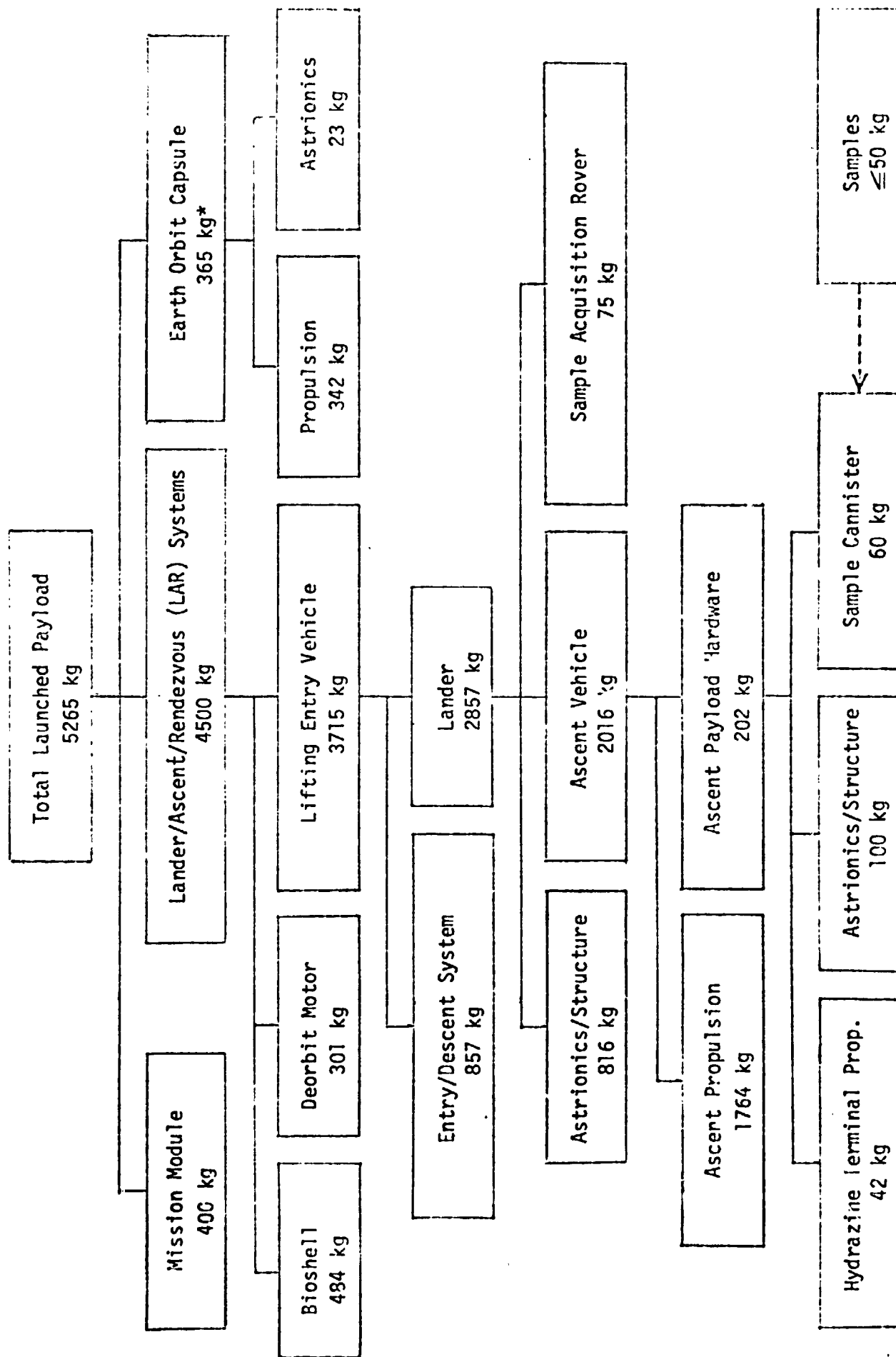


Table 22

SSEP/SAIL BASELINE MISSION DESCRIPTION

MISSION Mercury Orbiter w/Rough Landers

OBJECTIVES To obtain a global geologic map of the planet, to study planet interaction with interplanetary fields and plasma, and to conduct initial in situ surface investigations at 3-4 selected sites.

SPACECRAFT New, based on LPO design base; axis stabilized; Nadir pointing; low alt. thermal balance; net mass of ~ 600 kg^a.

PROBE(S) JPL Alternate Lander design; 3-4 units; gross mass allowance of ~ 200 kg/unit^b.

MISSION PARAMETERS

LAUNCH PERIOD: 1987-1989, ARRIVAL TIME: preferably near aphelion

SPACECRAFT STAYTIME: >180 days, PROBE(S) LIFETIME: >30 days

ORBIT PARAMETERS:DEFINITIONFUNCTION

- | | |
|---------------------------------|-------------------|
| 1) 500 km alt., circular, polar | global mapping |
| 2) 50x500 km alt., polar | lander deployment |

DEPLOYMENTS: Landers deployed from 50x500 km alt. orbit; orbiter deployment cycle ΔV requirements = 235 mps/lander

COMMENTS

- a) net mass excludes all propulsion
- b) all-up mass added to orbiter bus; includes 3-stage solid retro for landing from 50x500 km altitude orbit.

SSEP/SAIL BASELINE MISSION DESCRIPTION

MISSION Multi-Asteroid Survey w/Penetrators

OBJECTIVES To rendezvous with and globally map at least three asteroids of different classes; Vesta, a C-type, and a S-type asteroid are prime targets; a penetrator is deployed at each target; complete mapping at ≤ 100 m resolution, and detailed mapping at ≤ 10 m resolution are desired

SPACECRAFT New, but drawing heavily on JO and LPO design bases; axis-stabilized; net mass allowance of ~ 500 kg^a

PROBE(S) Mars penetrator design with larger tube-launched retro and cruise A/C capability; gross mass allowance of ~ 75 kg/unit^b; one unit per target plus one spare desired.

MISSION PARAMETERS

LAUNCH PERIOD: 1988-1990, ARRIVAL TIME: not between -90 and +30 days of Earth-Target conjunction

SPACECRAFT STAYTIME: 60-90 days; PROBE(S) LIFETIME: 30 days

ORBIT PARAMETERS:

<u>DEFINITION</u>	<u>FUNCTION</u>
1) 800 km alt., circular, polar	initial global mapping
2) 25-50 km alt., circular, polar	penetrator deployment and detailed mapping

DEPLOYMENTS: Penetrators tube-launched at 150 mps from 25-50 km alt. orbit; additional bus ΔV may be required.

COMMENTS a) net mass excludes all propulsion
b) all-up mass added to spacecraft bus

Table 24
SSEP/SAIL BASELINE MISSION DESCRIPTION

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MISSION Encke Sample^a Return

OBJECTIVES To conduct a thorough investigation of the comet over at least 50% of its orbital motion including perihelion passage, and to return samples of the nucleus to the earth; science objectives similar to Halley rendezvous mission but enhanced by sample return capability.

SPACECRAFT Mission module integrated with transport vehicle; MSSR design base; includes soft-docking capability; ~400 kg.

PROBE(S)^b Lander/Ascent/Rendezvous (LAR) Probe \pm 500 kg; Surface Base Station \pm 100 kg; Return Capsule (excluding sample^a) \pm 330 kg.

MISSION PARAMETERS

LAUNCH PERIOD: 1990-92^c, ARRIVAL TIME: <2 yrs before perihelion^d

SPACECRAFT STAYTIME: \geq 90 days^e, PROBE(S) LIFETIME: \geq 550 days

CIRCUMNAVIGATIONS: TBD; as required by transport system to map nucleus before LAR descent, and to meet coma science objectives.

STATIONKEEPING: TBD; as required by remote sensing and coma science requirements.

DEPLOYMENTS: SSEP/SAIL - as required by LAR descent requirements
LAR - as required by base station requirements

RECOVERIES: SSEP/SAIL - passive attitude control for LAR docking
LAR - active rendezvous after multiple site sampling of nucleus.

EARTH RETURN ORBIT: 175 nm circular orbit for direct Shuttle recovery

COMMENTS

- a) \leq 25 kg sample
- b) maximum mass on board transport system
- c) 1994 apparition
- d) permits half-orbit staytime of deployed base station
- e) minimum staytime for multiple sample acquisition and recovery by LAR.

Table 25

DERIVED SEP MISSION PARAMETERS

PROPULSION:

Launch Vehicle
Escape System (IUS stages)
SEP Transport System
Earth-Orbiter System

Orbit (rendezvous)
SEP Spiral
Chemical Retro

Direct vs. Satellite-Assist
Earth-Storable vs. Space-Storable vs. Solid

MASS PERFORMANCE:

Injected Mass (launch C3)
SEP Module Mass
SEP Propellant Mass
Chemical Retro Inert Mass (hyperbolic approach speed)
Retro Propellant Mass
Mass Margins

TIMES:

Launch Date
Launch Window
SEP Propulsion Time
Transit Times
Spiral Times
Encounter Date(s)

PROFILES:

Launch Sequence
Approach Events
Transfer Events
Encounter Events*
Return Events*

*As applicable prior to jettison

Table 26

DERIVED SAIL MISSION PARAMETERS

PROPULSION:

Launch Vehicle
Escape System (IUS vs SAIL spiral)
SAIL Transport System
Encounter System

None (rendezvous)
SAIL Spiral
Chemical Retro

Impulsive vs. Satellite-Assist
Earth-Storable vs. Space-Storable vs. Solid

MASS PERFORMANCE:

Injected Mass (launch C3)
SAIL Module Mass (characteristic acceleration at 1 AU)
Chemical Retro Inert Mass (hyperbolic approach speed)
Retro Propellant Mass
Mass Margins

TIMES:

Launch Date
Launch Window
SAIL Time
Transit Time
Spiral Times
Encounter Date(s)

PROFILES:

Launch Sequence
Approach Events
Transfer Events
Encounter Events*
Return Events*

*As applicable prior to jettison

Launch Window

Launch window is the period of time (usually measured in days) required to assure Shuttle-launch of the mission(s) within a given opportunity. For scheduling and associated cost reasons, the larger the available launch window the better. This provides the flexibility to fit into potentially high-traffic launch periods anticipated with the STS without serious mission impacts.

Outbound Flight Time

Outbound flight time is a free parameter derived by combining the baseline mission description with a specific design concept. Obviously, the shorter the flight time (again, all other things equal) the better.

Return Flight Time

Return flight time is also a free parameter of the two sample return missions which is needed to determine total mission time. To the degree that return flight time shortens total trip time, it should be as short as possible.

Arrival Time

Many of the baseline mission definitions provide some indication of desired arrival times. Significant departures in arrival time from these guidelines due to performance or design constraints would be grounds for discrimination against the system in question. For example, arrival times for the MSSR mission might be a source of discrimination if associated Mars weather conditions (e.g., dust storms) preclude an early landing.

Stay Time

Stay time applies to both the Mars and Encke Sample Return Missions and the Multi-Asteroid Rendezvous encounters. In general, it is anticipated that overall performance will be adversely affected by long stay times. The system which can provide the longest stay time (within limits) within comparable total flight times would receive preferential consideration.

Sensitivity to Increased Payloads

Since the baseline mission descriptions are only, at best, forecasts of anticipated mission configurations, it is important to understand the depth of performance each delivery system possesses. Although it's not clear over what range in payloads this sensitivity should be measured, a second performance benchmark for somewhere between 10% and 50% increased payloads (the specific value probably being mission dependent) may be requested.

Sensitivity to Transport System Degradation

The effect of system performance degradations on mission characteristics is to be determined. Of particular concern for SSEP are: a) reduction in power conversion efficiency, b) loss in engine performance, and c) whole engine losses. For SAIL, a) degradation in reflectivity, and b) effective sail area due to meteorite damage, should be considered.

B. SCIENCE DISCRIMINATORS

Cruise Science Interference

A brief investigation of probable interference of the systems designs and their operations with traditional field/particle cruise science instruments is desired since some of the transfer flight profiles will be in regions of new interplanetary interest; e.g. exoclyptic regions enroute to Halley, and spiral capture at Mars.

Encounter Science Interference

The specific question of concern is: "Will separate deployable payloads be necessary during encounter to achieve remote sensing objectives at asteroid rendezvous and on an Encke Sample Return mission?" If so, what additional system capabilities are implied for this capability. Obviously, the less the better.

Viewing Constraints

If encounter science can be performed without deployment of the remote sensing science payload (see B2 above), what are the viewing constraints and how do they impact the science payload?

Attitude Stability

Again, if encounter science does not require temporary payload deployment (as in B2 above), is the transport system stability adequate for science objectives, or is an isolated science platform required, or is there measurement degradation? Quantitative responses to these questions will be necessary even to make qualitative judgements of the consequences.

C. SPACECRAFT DISCRIMINATORS

Required Power/Command Support

Two subsystems, Power and Command, conceivably could be required to support the operations of either the SSEP or SAIL

transport systems, either periodically or continuously. A comparison of the demands each transport system adds to it's payload is a potential discriminator.

Provided Power Support

The SSEP has the potential for providing power to its payload prior to jettisoning or continuously if the payload is not jettisoned, e.g., the Multi-Asteroid Rendezvous mission. The extent to which this capability simplifies or otherwise benefits the payload compared to a similar SAIL payload is a discriminator.

Communications Constraints

Both the SSEP and SAIL concepts will likely rely on their payloads to provide the communications link to earth for their command/control. The degree to which this impacts the design and operation of the payloads could be a discriminator, as well as any constraints imposed by the transport systems on otherwise routine payload communications during the mission cruise phase.

Viewing Constraints

For missions which either retain their payloads or must recover a portion of the deployed payload, i.e. the Multi-Asteroid Rendezvous, Mars Sample Return, and Encke Sample Return missions, the viewing constraints imposed by the transport system on the spacecraft during encounter operations are a discriminator. For example, how are viewing conditions inhibited (if at all) during terminal rendezvous and docking?

Attitude Stability

Both attitude stability and attitude constraints may invoke significant penalties on the spacecraft of the three missions just mentioned above. The degree of constraint and resultant design modifications are possible discriminators.

Thermal Control Impact

The concern here is related to the shadowing of either the sun or deep space by the transport system, thereby creating new thermal control problems for the payload. This may be a somewhat greater problem on jettisoned missions where payloads could conceivably be required to operate in two very different thermal regimes. If the impact causes additional thermal control design for the payload of one system, but not the other, it's a discriminator.

Supporting Chemical Propulsion

On all missions where spacecraft propulsion is required in addition to the transport system, and the transport trajectory affects the size of that propulsion subsystem potential discrimination exists between the transport systems. As a guideline, the transport system which minimizes the post-jettison spacecraft propulsion requirements will be preferred (all other things being equal).

Assembly/Departure Constraints

Any constraints imposed on the payload as a result of transport system assembly/deployment and start-up requirements, i.e., undesirable attitude, communication black-outs, delayed stabilization, etc. could become discriminators in a comparison of earth departure sequences.

Target Approach Constraints

For comet missions specifically, the direction of encounter approach impacts both terminal rendezvous capability and spacecraft survivability (due to dust hazards). Approach paths of both transport systems to comet rendezvous should be reviewed for consistency with spacecraft design/operations requirements; conflicts would be sources of discrimination.

Maneuverability Constraints

Maneuverability constraints imposed by the large structures and attendant orientation requirements could considerably complicate spacecraft design requirements on the sample return and Multi-Asteroid missions. Discrimination in preference of one transport system over the other resulting from this consideration would be combined with discriminatory implications of viewing constraints and attitude stability which are closely related to maneuverability constraints.

Docking Loads Constraints

Docking loads tolerable by the waiting transport system applied to sample return missions are a discriminator if they differ significantly between transport systems and create docking design/operations constraints. This may also be an issue for the Multi-Asteroid mission if the spacecraft must temporarily free itself of the transport system to perform its encounter functions.

Electrical Charging

The potential for electrical charging created on the large transport structures during interplanetary cruise and resultant danger of arcing both in the spacecraft and transport subsystems should be briefly investigated. Obvious problems would likely be solved; but if not, would just as obviously become discriminators.

D. NAVIGATION

Viewing Constraints

When on-board tracking is required to insure acceptable encounter accuracy, star and target sightings will be required by the spacecraft. Viewing constraints imposed on these operations by transport system attitude requirements and/or physical obstructions are potential discriminators on mission navigation capability.

Attitude Stability

The inherent transport system stability impacts the sighting capability and accuracy of spacecraft sensors for navigation purposes. Design and/or accuracy compensations in navigation systems caused by transport system characteristics can cause discrimination between alternative transport options.

Operational Procedure

Operational procedures required to obtain acceptable tracking data for orbit determination should be reviewed to determine impacts on performance and/or spacecraft operations. Procedures which adversely affect performance (e.g., long coast periods) or spacecraft operations (e.g., attitude control during coast periods) are possible discriminators resulting from navigational requirements of the competing transport systems.

Accuracy

The end goal of navigation is to provide acceptable encounter accuracy for the achievement of the payload objectives. Having made all the necessary concessions, adjustments and procedural changes necessary to accomodate both transport and payload requirements, the resultant accuracy of the navigation is itself a discriminator between the competing transport systems. Accuracy should be judged against a priori requirements as well as comparative capabilities so that unnecessary discrimination doesn't occur.

The final step in defining the mission discriminators was the assignment of relevant discriminators to each of the six reference

planetary missions. This was done using the discriminator and baseline definitions and assuming a jettison or non-jettison mode for the low-thrust systems at encounter. The encounter configurations assumed are presented in Table 27. The matrix of discriminator assignments is presented in Table 28. It is seen from the totals at the bottom of the page that those missions which don't jettison their low-thrust transport system at encounter have a stronger interface between payload and propulsion and hence more mission elements for discrimination. The Encke Sample Return mission is the extreme case with all 28 discriminators being relevant contributors to utility comparison of the SSEP and SAIL systems.

Table 27

SSEP/SAIL ENCOUNTER CONFIGURATION GUIDELINES

MISSION	CONFIGURATION		
	Jettison	Non-Jettison	Deploy/Recovery
Halley Rendezvous w/Nucleus Probe	X		
Saturn Orbiter w/Titan Lander	X		
Mercury Orbiter w/Rough Landers	X		
Mars Surface Sample Return	X ^a		X ^b
Multi-Asteroid Rendezvous w/Penetrators		X	
Encke Rendezvous and Sample Return	X ^a		X ^b

- a. Jettison on returning earth approach
- b. Deploy lander and recover samples

Table 28

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	Halley Rendezvous	Saturn Orbiter	MSSR	Mercury Orbiter	Multi-Ast. Rendezvous	Encke SSR
<u>PERFORMANCE</u>						
Launch Vehicle	X	X	X	X	X	X
Launch Window	X	X	X	X	X	X
Outbound Flight Time	X	X	X	X	X	X
Return Flight Time			X			X
Arrival Time	X		?	X	X	X
Stay Time			X		X	X
Sensitivity to increased payloads	X	X	X	X	X	X
Sensitivity to transport system degradation	X	X	X	X	X	X
<u>SCIENCE</u>						
Cruise Science Interference	X	X	X	X	X	X
Encounter Science Interference					X	X
Viewing Constraints					X	X
Attitude Stability					X	X
<u>SPACECRAFT</u>						
Required Power/Command Support	X	X	X	X	X	X
Provided Power Support	X	X	X	X	X	X
Communications Constraints	X	X	X	X	X	X
Viewing Constraints			X		X	X
Attitude Stability			X		X	X
Thermal Control Impact	X	X	X	X	X	X
Supporting Chemical Propulsion	X	X	X	X	X	X
Assembly/Departure Constraints	X	X	X	X	X	X
Target Approach Constraints						X
Maneuverability Constraints			X		X	X
Docking Load Constraints					?	X
Electrical Charging	X	X		X	X	X
<u>NAVIGATION</u>						
Viewing Constraints	X		X	X	X	X
Attitude Stability	X	X	X	X	X	X
Operational Procedures	X	X	X	X	X	X
Accuracy	X	X	X	X	X	X
Total Assignments (max. of 28 possible)	18	17	23	18	25	28

3. Reports and Publications

Science Applications, Inc. is required, as part of its advanced studies contract with the Planetary Programs Division, to document the results of its analyses. This documentation traditionally has been in one of two forms. First, reports are prepared for each scheduled contract task. Second, publications are prepared by individual staff members on subjects within the contract tasks which are considered of general interest to the aerospace community. A bibliography of the reports and publications completed during the contract period 1 February 1976 through 31 January 1977 is presented below. Unless otherwise indicated, these documents are available to interested readers upon request.

3.1 Task Reports for NASA Contract NASW-2893

1. "Mercury Orbiter Transport Study," Report No. SAI-1-120-580-T6.
2. "Planetary Opportunities Calendar," Report No. SAI-1-120-580-T7.
3. "Galilean Satellite Compositional Measurement with Penetrators," Report No. SAI-1-120-580-S2, February 1977.
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3.2 Related Publications

1. "Planetary Exploration-Options for the Future," J.C. Niehoff and L.D. Friedman, AAS/AIAA, Bicentennial Symposium, Session I, October 1976.
2. "Round-Trip Mission Requirements for Asteroids 1976AA and 1973EC," Icarus (to be published).
3. "Penetrator Mission Concepts for Exploration of the Galilean Satellites," J.C. Niehoff, A.L. Friedlander, and D.R. Davis, AIAA Paper No. 76-800, AAS/AIAA Astrodynamics Conference, August 1976.
4. "Launch Opportunity Classification of VEGA and ΔV -EGA Trajectories to the Outer Planet," A.L. Friedlander, M.L. Stancati and D.F. Bender, AIAA Paper No. 76-797, AAS/AIAA Astrodynamics Conference, August 1976.
5. "Final Report and Recommendations", Ad Hoc Surface Penetrator Science Committee, NASA Headquarters, August 1976.

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